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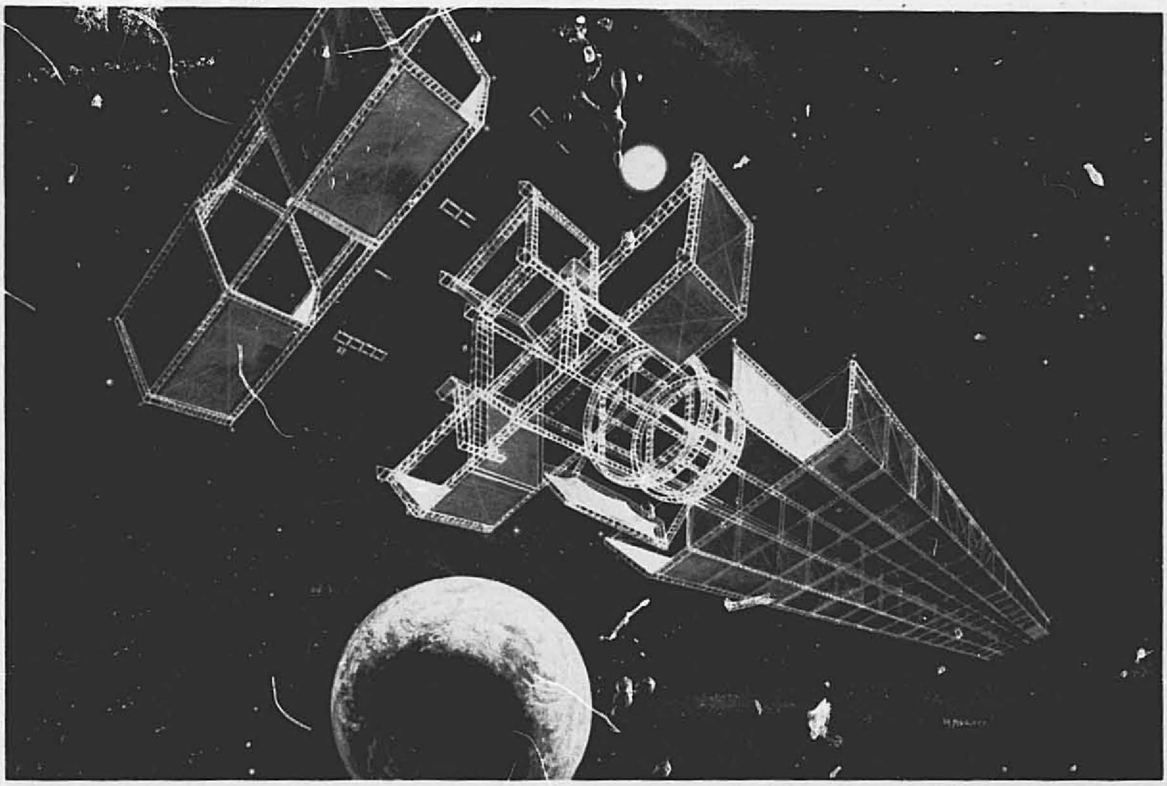

Satellite Power Systems (SPS) Concept Definition Study

FINAL REPORT

VOLUME V

TRANSPORTATION AND OPERATIONS ANALYSIS

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FOREWORD

This document, *Transportation and Operations Analysis*, is Volume V of the SPS Concept Definition Study (Contract NAS8-32475), Exhibits A and B, and also incorporates results of NASA/MSFC in-house effort. Other volumes of the final report that provide additional detail are listed below.

Volume	
I	Executive Summary
II	SPS System Requirements
III	SPS Concept Evolution
IV	SPS Point Design Definition
VI	SPS Technology Requirements and Verification
VII	SPS Program Plan and Economic Analysis

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1.0 INTRODUCTION

1.0 INTRODUCTION

Transportation and operations will play an integral role in the ultimate success of SPS. As these early studies of SPS feasibility progress, the significance of that role becomes increasingly clear. Satellite concepts have been completely reconfigured because of transportation systems selections or to facilitate otherwise unreasonable construction demands. The analyses leading to the choice of a GEO construction orbit using an electric COTV is a good example of the strong interrelationships among transportation systems, operations and costs. In essence, SPS poses an excellent challenge to the advanced systems engineer wherein the applicability of alternative hardware elements or operational processes must always be considered in the context of the overall SPS program.

The material contained in this volume should provide the reader with an insight into these relationships. Primary emphasis has been placed on the development of transportation systems concepts that operate in support of the construction operations required to assemble a 5-gigawatt satellite in space. Additionally, data supporting the concept for construction of a ground receiving antenna (rectenna) is presented. The interplay among the satellite, rectenna, transportation systems and operations can be more easily described by using the functional diagram shown in Figure 1.0-1. Starting with the point design satellite concept, alternative man-machine assembly processes were analyzed for each of the major satellite subsystems, e.g., structural frame fabrication, solar blanket installation, etc. In conducting these analyses, the HLLV payload and cargo bay dimensions had to be recognized as a constraint on material size and mass. The concepts for subsystems assembly had to be integrated and sequentially time-phased in order to develop a construction schedule. Estimates of crew sizes could then be made and their support requirements conceptualized. The totality of these concepts and requirements led to the definition of a candidate space construction base.

The construction operations and their sequences establish the time-phased cargo demands for transportation systems. Before a space logistics traffic model could be defined, however, a cargo packaging analysis had to be conducted. This involved the payload integration of hardware components from each of the subsystems while assuring that the cargo supply to the space construction base would meet or exceed the cargo demands for each subsystem. As might be anticipated, this analysis resulted in numerous iterations and modifications to the original construction schedule before compatibility was established.

Of the required transportation systems, only the capabilities of the Rockwell HTO-SSTO HLLV concept were assumed as limiting constraints. In a separate company-funded study, an analysis was conducted to further establish the design feasibility and performance capabilities of this HLLV concept. The promising results of that effort are reported on herein. The transportation systems concepts in concert with the space logistic traffic model set

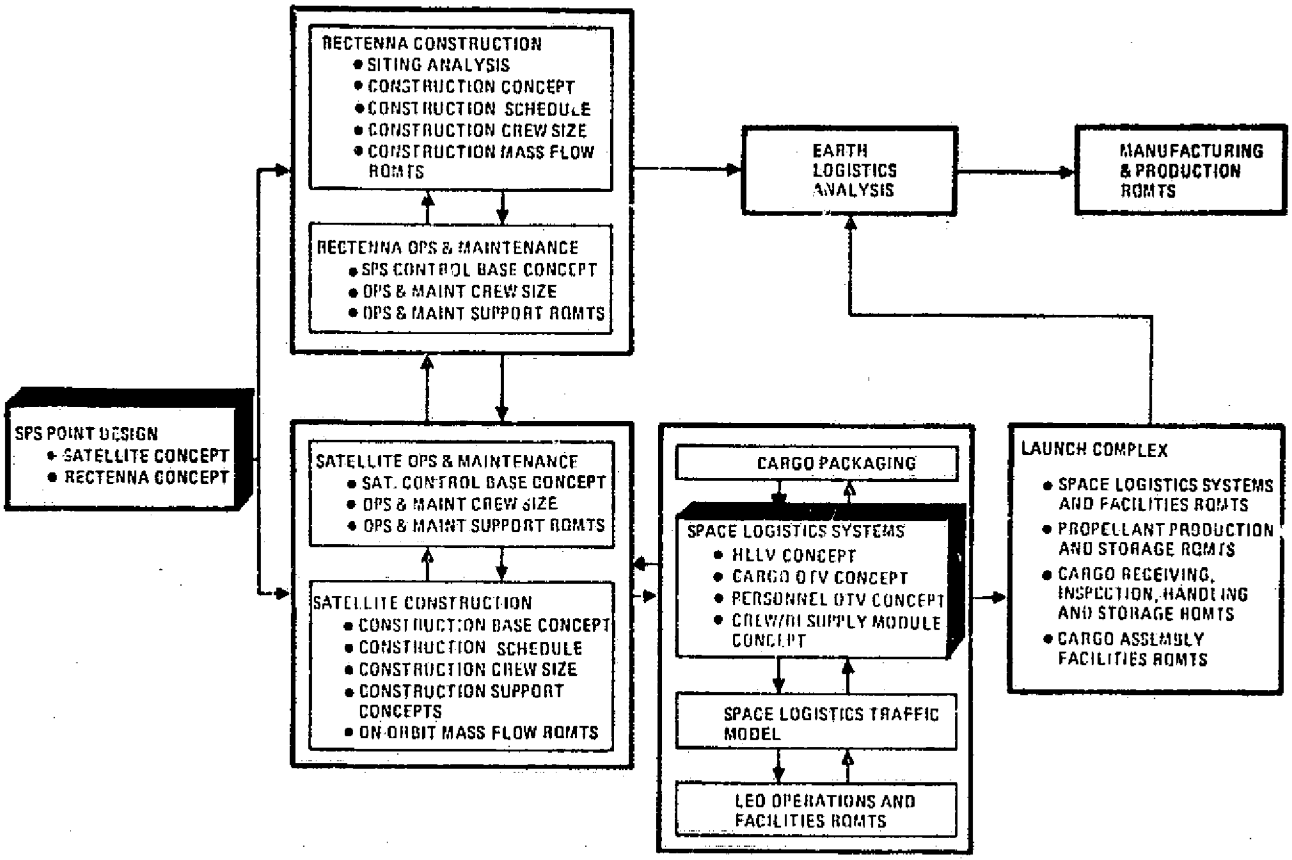

Figure 1.0-1. Systems and Operations Interrelationships

the operational requirements for the earth launch complex. These launch complex operational requirements were used as a basis for comparing a ballistic two-stage, vertically launched HLLV with the Rockwell concept.

Finally, the earth logistics demands imposed by the requirements at the earth launch complex and at a typical rectenna site were compiled. Of these two, the logistics demands at a rectenna site were found to be a daily factor of 20 over those at the earth launch complex even though a 12-month delivery schedule was estimated for the rectenna site versus a 3-month schedule for the earth launch complex. The results of all the above analyses led to the following general conclusion: There does not appear to be any technological or cost issues associated with the transportation systems or operational processes which should negate or delay further the conduct of more detailed investigations into these concepts. Indeed, the results achieved to date tend to reinforce the viability of SPS as an economically competitive and environmentally desirable source for future electrical power production.

Having resolved the issue of LEO versus GEO construction orbit in a previous analysis, the task remained for sizing the COTV. An electric COTV is quite modular in the sense of employing numerous thrusters and power processors, thus it can be sized very small or a concept can be designed to transport the entire cargo required to construct a single satellite. On the one hand, small COTV's - although very versatile - would create the demand for a large fleet complement and the requirement for continuous command and control of many in-flight vehicles. The other extreme presents an "all-the-eggs-in-one-basket" syndrome associated with a single, very large COTV which cannot depart for GEO until all the cargo has been delivered to LEO from earth. A compromise concept between these extremes was arrived at based upon considerations for fleet size, LEO-to-GEO up-down trip times, HLLV payload capabilities, and satellite construction schedules.

2.0 TRANSPORTATION SYSTEMS ANALYSIS

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A key element in the overall feasibility of the Satellite Power System (SPS) concept is the transportation concept(s), either available or projected to be available in the time frame being considered. Since the transportation costs contribute significantly to the total SPS life-cycle cost, methods of reducing transportation cost and/or simplifying transportation operations greatly enhance the acceptance of the SPS concept.

The major elements of the SPS transportation system consist of:

- Heavy-Lift Launch Vehicle (HLLV) - Cargo to LEO
- Space Shuttle Transportation System (SST) - Personnel to LEO
- Cargo Orbit Transfer Vehicle (COTV) - Cargo from LEO and GEO
- Personnel Orbit Transfer Vehicle (POTV) - Personnel from LEO to GEO
- Intra Orbit Transfer Vehicle (IOTV) - On-orbit cargo transfer

These transportation concepts embody feasibility issues of their own; however, these have not been addressed other than in a recognitive sense and, where available, data from previous studies have been utilized.

The overall evolution of the SPS transportation system requirements are depicted in Figure 2.0-1. The baseline Space Shuttle is utilized to satisfy the SPS technology development issues requiring space flight. Shuttle derivatives are employed for the construction of the SPS 1-GW prototype satellite. The SPS operational transportation system must then be employed to satisfy the high mass flow requirements to expand the prototype capability to the operational 5-GW configuration. This report is devoted primarily to the SPS operational transportation system elements and trade studies.

Trade studies were conducted on HLLV type (ballistic or winged), SPS construction site (LEO or GEO), and COTV concepts (chemical, nuclear, and electric self-powered or dedicated). A preferred system concept was selected on the basis of cost, operational complexity, and environmental considerations.

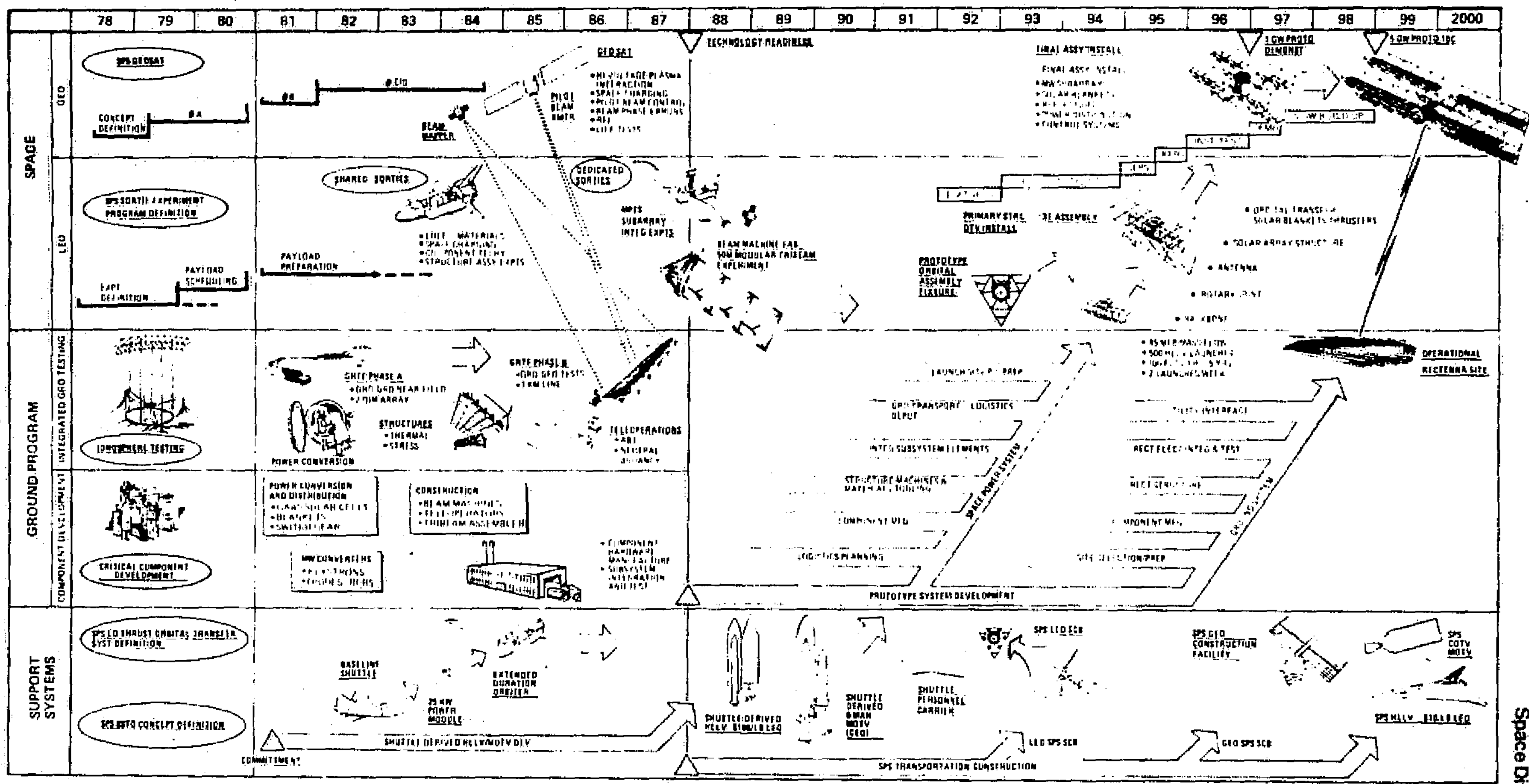

Figure 2.0-1. Overall Transportation Requirements

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3.0 TRANSPORTATION SYSTEM OPTIONS AND REQUIREMENTS

3.0 TRANSPORTATION SYSTEM OPTIONS AND REQUIREMENTS

Basic operational transportation system requirements are summarized in Table 3.0-1.

Table 3.0-1. Operational Transportation System Requirements

✓ ORBITAL REQUIREMENTS

- LEO - 500 KM @ 28.5° & 550 KM EQUATORIAL
- GEO - ~35,800 KM EQUATORIAL

✓ MASS FLOW TO ORBIT

- 200 - 350 x 10^6 KG/YR
- 500 - 4000 FLTS/YR

✓ ENVIRONMENTAL CONSIDERATIONS

- FUELS/FUEL STORAGE
- ATMOSPHERIC CONTAMINATION
- ACOUSTIC LEVELS
- ORBITAL DEBRIS (DOWN PAYLOAD)

✓ COST

- OPERATIONS
- RECOVERY/REUSABILITY/ATTRITION
- TURNAROUND TIME

✓ SCHEDULE

- EXISTING TRANSPORTATION SYSTEMS
- SYSTEMS PROJECTED FOR THE LATE 1990'S AND BEYOND

Two low earth orbits were considered: a 500-km orbital altitude at approximately 28.5 degrees inclination, and at equatorial LEO of 550 km. The desired LEO is a function of earth launch vehicle concept selection.

The minimum mass flow to orbit for a mature program ranges from 200 to 356×10^6 kg/year for the electric OTV concept, and as high as 10^9 kg/year for a chemical OTV. Dependent upon HLLV payload capability, this results in HLLV flight rates ranging from 500 to 4000 per year.

Because of the large quantities of fuels consumed by the HLLV, environmental consideration must be given to fuel consumption and storage hazards, atmospheric contamination in localized areas, and the maximum allowable acoustic levels in the area of the launch site. In addition, since a significant mass will be required for packaging of payload elements, the ultimate disposition of this mass in orbit must be considered.

Since the operational cost of the transportation system represents a major portion of SPS costs, transportation system selection is dominated by dollars/kg to orbit.

Primary drivers in establishing transportation system requirements are the high density mass flow requirements to LEO and GEO. Another major factor, within the transportation system itself, is the COTV concept employed for transfer from LEO to GEO. Dependent upon COTV concept selection, two of every three HLLV flights to LEO may be required for OTV propellant resupply.

Typical mass-to-orbit requirements are depicted in Figure 3.0-1 for various elements and options. It is noted that the upper curves include OTV maintenance masses only, and do not include initial OTV structural dry weight. The large difference in mass-to-orbit requirement for chemical as opposed to electric OTV reflects the more favorable orbital burden factor achievable with an argon ion electric propulsion system.

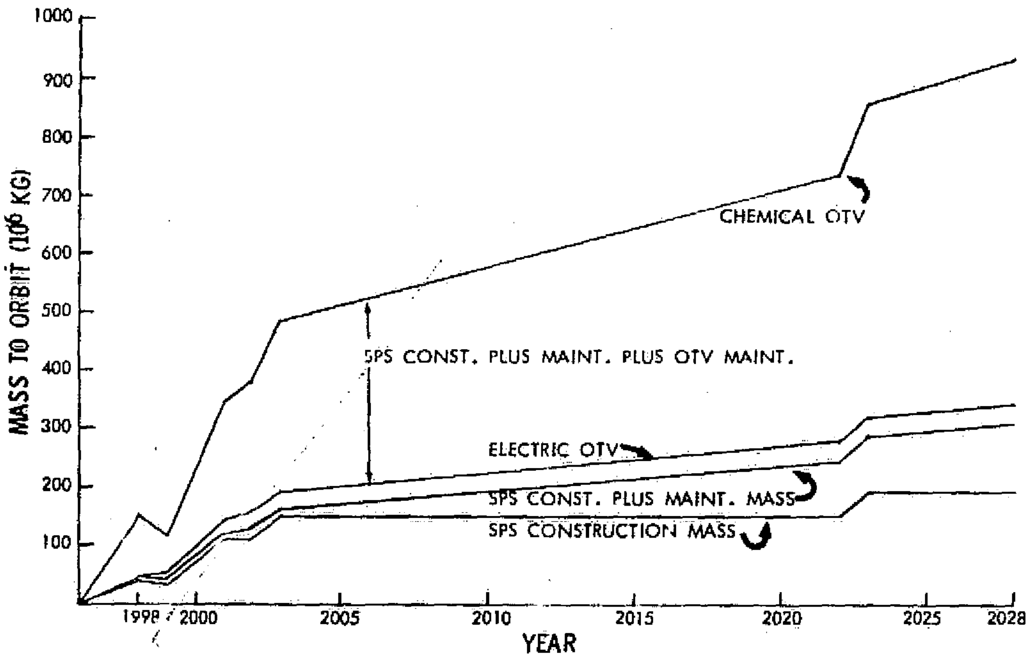

Figure 3.0-1. Mass-to-Orbit Requirements

The various transportation system elements considered are summarized in Table 3.0-2 and Figure 3.0-2.

Table 3.0-2. Transportation System Options

CARGO	PERSONNEL/PRIORITY CARGO
EARTH - LEO	
• TWO-STAGE BALLISTIC • TWO-STAGE WINGED - VTO • HTO - SSTO • HTO - TWO STAGE	• SHUTTLE DERIVATIVE • HTO-SSTO • HTO - TWO STAGE
LEO - GEO	

• CHEMICAL
• NUCLEAR - GCR
• ELECTRIC
  • SELF PROPELLED
  • DEDICATED
	• CHEMICAL
	• NUCLEAR - GCR

ON ORBIT	
• CHEMICAL	• CHEMICAL

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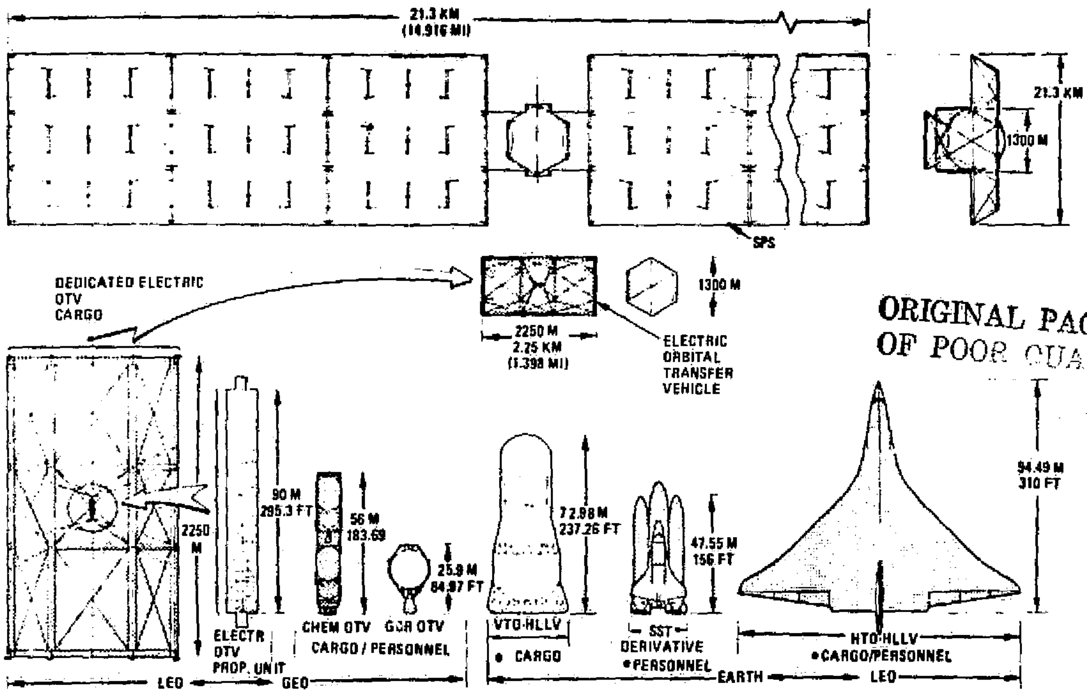


Figure 3.0-2. Transportation System Options -
Vehicle Size Comparisons

The vertical launch configuration, two-stage ballistic, is unmanned and utilizes a Space Shuttle derivative for personnel and priority cargo delivery to LEO. The one- or two-stage horizontal takeoff vehicle is manned and capable of both cargo and personnel delivery. Of the three OTV options studied for LEO-GEO transfer, the chemical and nuclear gas core reactors are capable of both cargo and personnel delivery. A chemical or nuclear OTV would be used for personnel/priority cargo delivery for the two electric cargo orbital transfer options.

The only option analyzed for on-orbit propulsion is a LOX/LH_2 chemical system.

4.0 EARTH-TO-LEO TRANSPORTATION SYSTEMS

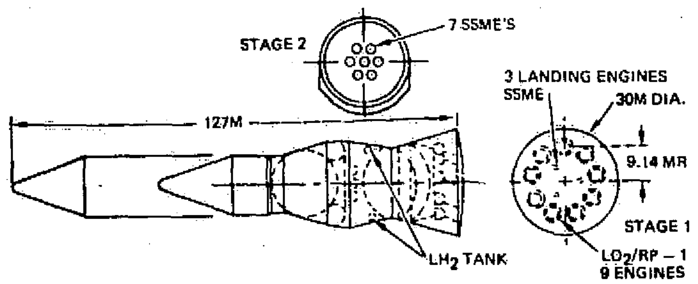
4.0 EARTH-TO-LEO TRANSPORTATION SYSTEMS

The two concepts selected for analysis were the two-stage ballistic system and a winged horizontal takeoff single-stage-to-orbit system. These represent a wide range of alternatives and both are economically competitive.

4.1 TWO-STAGE BALLISTIC HLLV

Two classes of vertical-launch HLLV's were selected for analysis, as illustrated in Figure 4.1-1, the smaller having a net payload capability of 91,000 kg and the larger, a net payload capability of approximately 400,000 kg. The 454,000-kg payload indicated in the figure includes the payload shroud.

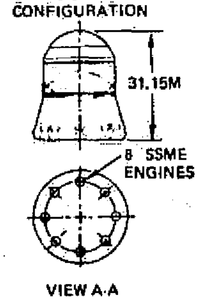
SMALL HLLV*

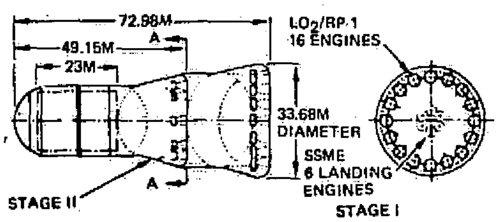


PAYLOAD	91 X 10^3 KG	(200 X 10^3 LBS)	WEIGHT
STAGE 1	30 M	(98 FT)	DIAM
	21 M	(69 FT)	LENGTH
	526 X 10^3 KG	(1.16 X 10^6 LBS)	LANDING WT
STAGE 2	27 M	(88 FT)	DIAM
	31 M	(102 FT)	LENGTH
	217 X 10^3 KG	(478 X 10^3 LBS)	LANDING WT

LARGE HLLV*

PAYLOAD	454 X 10^3 KG	(1 X 10^6 LBS)	WEIGHT
STAGE 1	34 M	(110 FT)	DIAM
	24 M	(78 FT)	LENGTH
	787 X 10^3 KG	(1.73 X 10^6 LBS)	LANDING WT
STAGE 2	28 M	(90 FT)	DIAM
	31 M	(101 FT)	LENGTH
	358 X 10^3 KG	(790 X 10^3 LBS)	LANDING WT





*DRAWINGS FROM D 180-20689 3 PART I VOL III BOEING SPS SYSTEM DEFINITION STUDY

Figure 4.1-1. Ballistic HLLV Configurations

The two-stage ballistic vehicle systems selected for study are derivatives of the vehicle concept described in the study performed by Boeing Aerospace Company, *Systems Concepts for SPS-Derived Heavy-Lift Launch Vehicles* (NAS9-14710), dated September, 1976. This vehicle represents a competitive class of vehicles that is recovered by way of a ballistic entry through the earth's atmosphere with parachutes and/or retrorockets providing a soft landing.

4.1.1 VEHICLE CONFIGURATION

The first-stage of the larger vehicle shown in Figure 4.1-1 is powered by 16 LOX/RPI gas generator cycle engines, each having a vacuum thrust of 9.06×10^6 N. Six SSME engines provide the necessary thrust for landing at sea. Thermal protection for the base area is achieved by using a metallic heat shield that is actively cooled, using a water spray during both ascent and descent.

The upper stage is powered by eight SSME engines having expansion ratios of 77.5:1. Each engine produces a vacuum thrust of 2.091×10^6 N and a specific impulse of 455.2 s. Base thermal protection is provided using a water spray similar to the booster. The landing system, however, employs a combination of retrorockets and parachutes. The payload is transported inside a shroud that can be collapsed for a shortened reentry configuration.

An in-depth analysis of the smaller HLLV configuration was dropped due to operational considerations discussed in a subsequent section of this report. The ballistic vehicle selected for cargo transportation has a delivery capability of approximately 400,000 kg to a 500-km orbit inclined 28.8 degrees to the equator. Although this particular size may not be optimum from a cost reference, it is felt to be an acceptable compromise between launch vehicle size and launch rate.

It is essential to develop transportation concepts that require minimum amounts of expendable hardware and refurbishment/repair effort. This fully reusable concept employs minimum maintenance philosophy in order to maintain costs at an acceptably low level.

4.1.2 MASS PROPERTIES

Weight data for the booster and upper stage are given in Tables 4.1-1 and 4.1-2, respectively. The gross liftoff weight of approximately 10^7 kg along with the previously described propulsion system characteristics yield a thrust-to-weight ratio of 1.3.

4.1.3 LAUNCH WINDOWS

Since several HLLV launches per day are required in support of the SPS, an analysis was conducted to determine the effect of launch windows on HLLV payload capability. Launch windows were developed for inclinations of 28.5 and 55 degrees, assuming launch from KSC to a 500-km orbit. Figure 4.1-2 presents the results of this analysis. As shown, the payload degradation is extremely sensitive to launch time and is more severe at the higher inclination. Only one in-plane launch opportunity occurs per day for the 28.5-degree inclination. For inclinations greater than 28.5 degrees, two in-plane launch

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Table 4.1-1. Ballistic HLLV Booster Mass

STAGE ELEMENT	10^3 kg	10^3 lbm
STRUCTURE	283.65	625.34
THERMAL PROTECTION SYSTEM	44.47	98.04
MAIN PROPULSION	177.75	391.88
AUXILIARY PROPULSION, RCS	1.49	3.28
LANDING AND AUXILIARY SYSTEM	30.48	67.19
PRIME POWER	0.74	1.62
ELECTRIC CONVERSION AND DISTRIBUTION	3.32	7.31
HYDRAULIC CONVERSION AND DISTRIBUTION	9.87	21.77
AVIONICS	2.43	5.36
ENVIRONMENTAL CONTROL SYSTEM	5.22	11.51
MASS GROWTH (10%)	55.94	123.33
DRY MASS (INCLUDING H_2O FOR TPS)	615.36	1,356.63
RESIDUAL AND UNUSABLE PROPELLANT	117.81	259.72
RESERVE RETRO PROPELLANT	6.97	15.37
USABLE RCS PROPELLANT	3.15	6.94
USABLE RETRO PROPELLANT	44.40	97.87
TOTAL INERT	787.69	1,736.53
ASCENT PROPELLANT	7,455.70	16,436.84
BLOW	8,243.39	18,173.37

Table 4.1-2. Ballistic HLLV Second-Stage Mass and Mass Sequence

DRY MASS	
STAGE ELEMENT	10^3 kg
STRUCTURE	155.43
THERMAL PROTECTION SYSTEM	3.30
MAIN PROPULSION	29.85
AUXILIARY PROPULSION	5.15
PRIME POWER	0.48
ELECTRIC CONVERSION AND DISTRIBUTION	0.68
HYDRAULIC CONVERSION AND DISTRIBUTION	3.59
AVIONICS	1.59
ENVIRONMENTAL CONTROL SYSTEM	2.07
CARGO SHROUD	33.01
PAYLOAD SUPPORT SYSTEM	1.27
GROWTH	22.40
DRY MASS	258.82

SECOND STAGE SEQUENCE	
EVENT	MASS AFTER EVENT 10^3 kg
STAGE AT MECO	749.58
ΔV RESERVES	736.63
APOGEE CIRCULARIZATION (OMS BURN)	719.11
RCS TRIM BURN	714.76
OMS TRIM BURN	713.06
DEPLOY PAYLOAD (MASS = 391,460 kg)	321.60
DEORBIT ΔV	313.14
H_2O EXPENDED DURING ENTRY	301.12
LANDING RETRO	279.85
MASS AT LANDING	279.85
RESIDUALS AND UNUSABLES	14.28
RESERVED LANDING PROPELLANT AND H_2O	6.75
DRY MASS	258.82

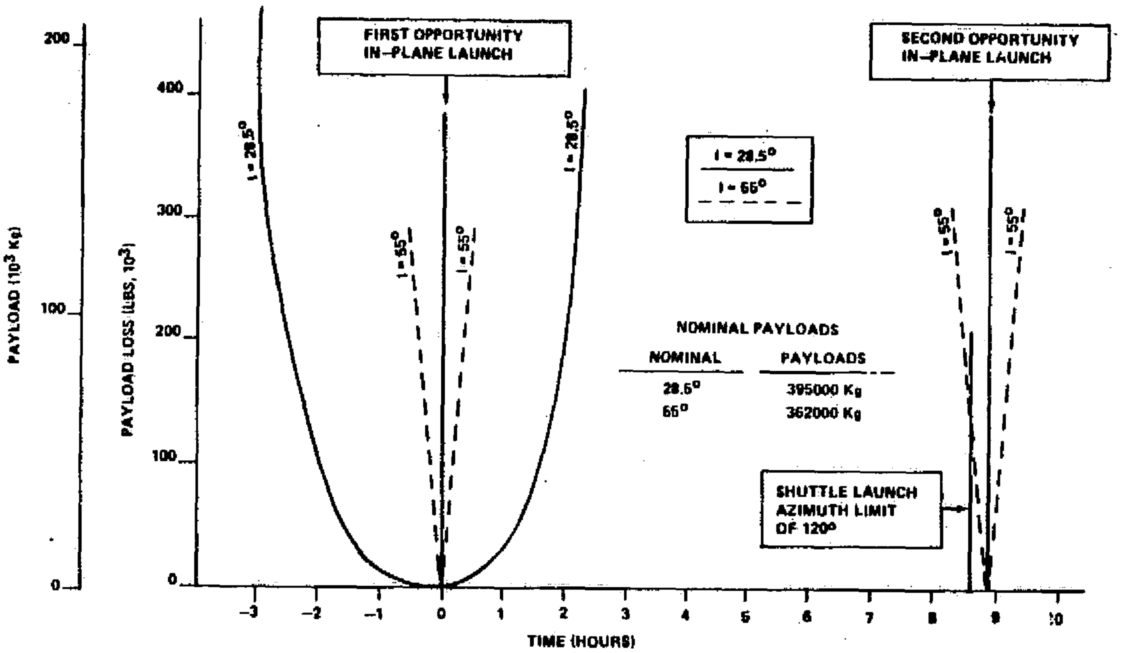

Figure 4.1-2. Payload Loss as a Function of Time of Launch

opportunities occur per day (assuming no launch azimuth restrictions) and as the inclination increases, the time between these opportunities increases. For the 55-degree inclination, the second coplanar launch opportunity occurs approximately 8.9 hours after the first opportunity; however, when launch azimuth limitations are imposed, this in-plane opportunity is negated.

4.1.4 FLIGHT PROFILE AND PERFORMANCE

A typical flight profile for the two-stage ballistic HLLV is described next. The vehicle is launched from Kennedy Space Center (KSC), with first-stage burnout and separation occuring at an altitude of approximately 65 km and a velocity of approximately 3000 m/s. The first-stage reenters and is recovered at sea for subsequent reuse. Second-stage main engines cut off at a perigee altitude of 93 km, with the required velocity to place the stage and payload at the desired apogee altitude. At apogee, the orbital maneuvering system (OMS) circularizes the stage and payload into the desired orbit. The payload is deployed and the second-stage deorbits (approximately 24 hours after launch) using the OMS, and reenters for subsequent recovery and reuse.

Performance analysis trade studies were conducted to determine the sensitivity of the HLLV payload capability to variations in orbital inclination and orbital altitude, assuming the launch site at KSC. Also, the HLLV payload capability sensitivity to launch site latitude was investigated.

Figure 4.1-3 shows payload capability to a 500-km orbit as a function of orbital inclination. As shown, the payload capability for a due-east launch (28.5-degree inclination) is 395,000 kg. The HLLV payload capability decreases to 362,000 kg for a 55-degree inclination. Also depicted in this figure are the current Shuttle launch azimuth constraints for a KSC launch.

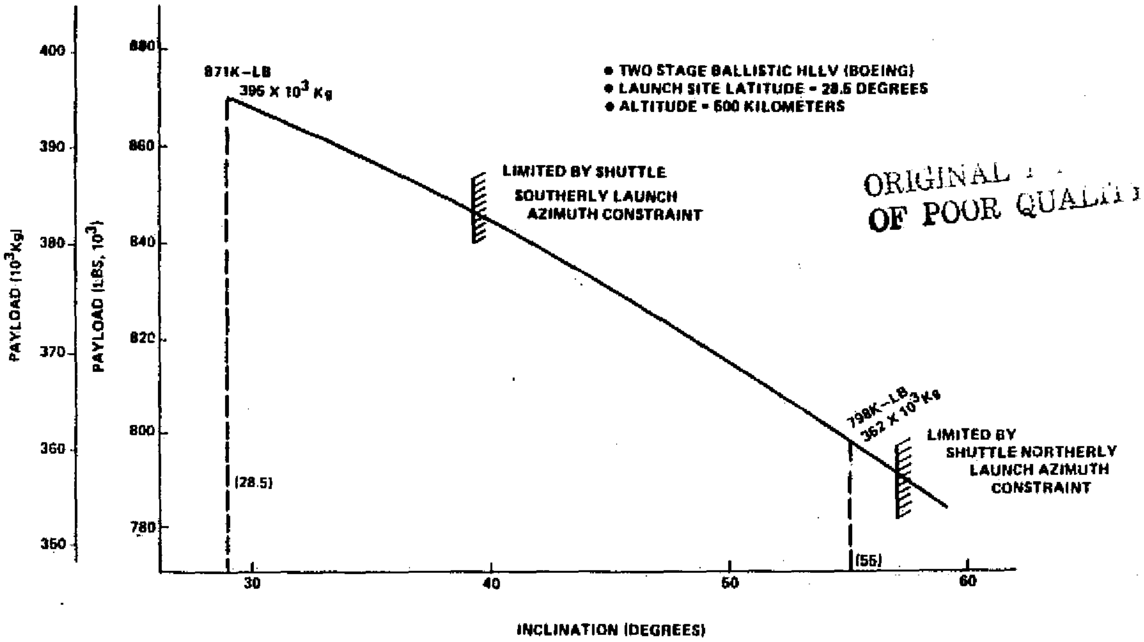

Figure 4.1-3. Payload Variation with Orbital Inclination

Figure 4.1-4 presents payload capability as a function of orbital altitude. The HLLV is launched from KSC on a 90-degree launch azimuth which yields a 28.5-degree inclination. The change in payload as a function of altitude change is approximately 113 kg/km.

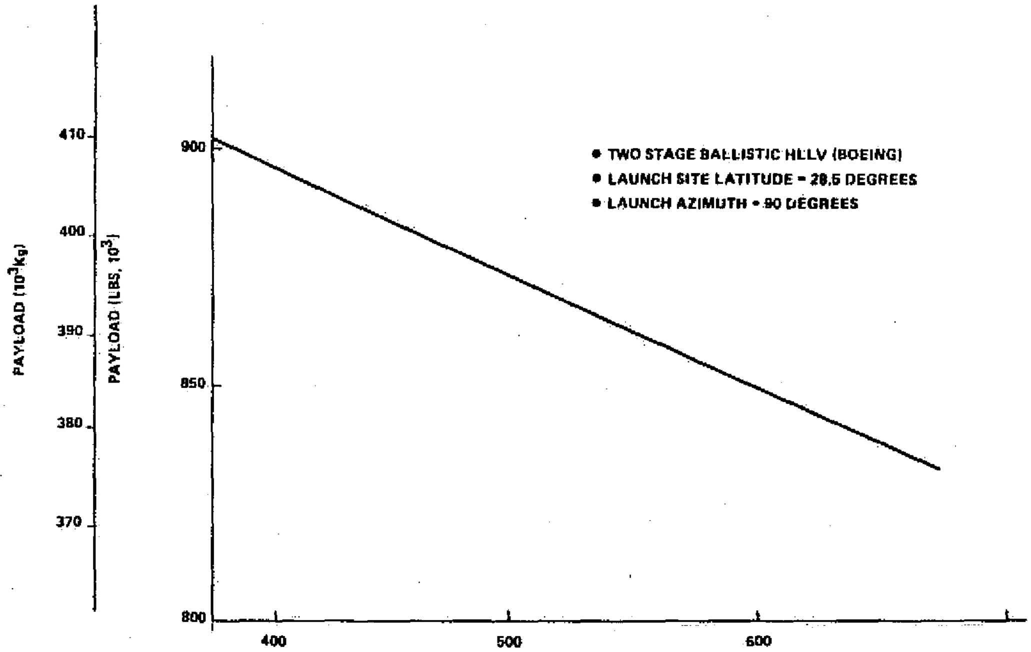

Figure 4.1-4. Payload Variation with Orbital Altitude

The baseline launch site selected for the SPS study was KSC; however, a trade study was conducted to determine the effect on HLLV payload capability of various launch site latitudes. Figure 4.1-5 presents the results of this analysis. Data presented are based on a 90-degree launch azimuth and a 500-km payload delivery altitude. An equatorial launch site would provide approximately a 4-percent increase in HLLV payload capability as compared to launch from KSC.

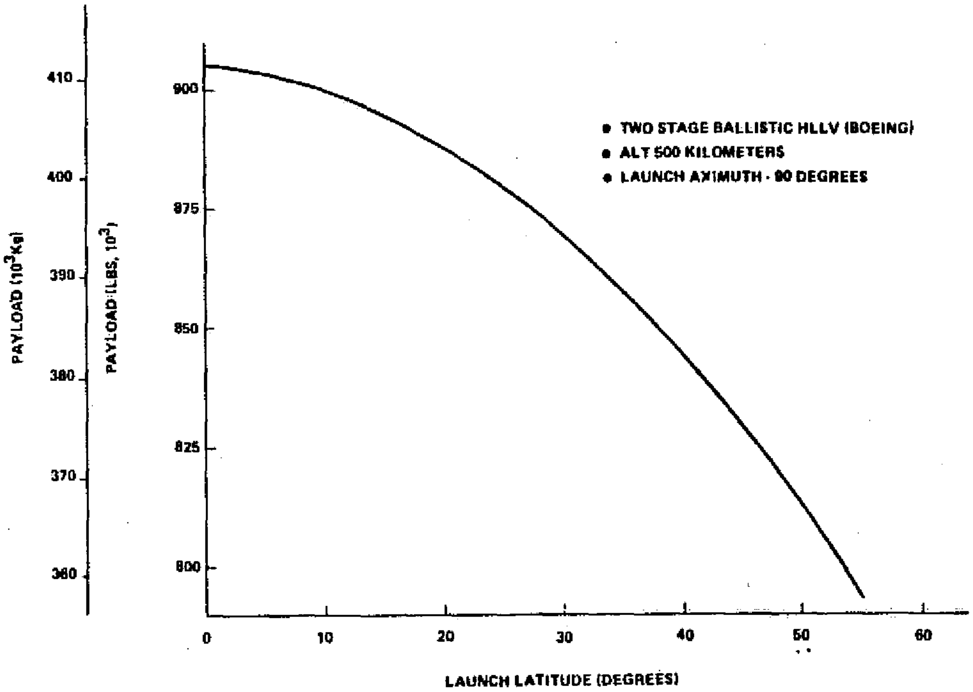

Figure 4.1-5. Payload Variation with Launch Site Latitude

4.2 HORIZONTAL-TAKEOFF HLLV - WINGED

Because the single-stage-to-orbit concept described herein was developed under Rockwell IR&D funding, a limited description only is presented.

The vehicle utilizes a wet-wing concept and multi-cycle airbreathing engines (turbojet/ramjet) from takeoff to ~M = 7. Three SSME-type engines are employed from M = 6 to LEO. The vehicle has a cargo bay 6×6×30 m, and is capable of placing 91,000 kg in a 550-km equatorial orbit.

The vehicle will take off from KSC, climb to 20,000 ft altitude, and cruise to the equator under turbojet power. After turning into the equatorial plane, the vehicle will begin its ascent under augmented turbojet power and transition to ramjet mode at approximately M = 3. At approximately M = 6, the SSME type engines will be ignited and throttled to maximum power while throttling down the ramjet engines and closing of the variable inlet. During reentry, the variable inlet ramp will be reopened and the vehicle will cruise back to the launch site on the airbreathing engine system.

4.2.1 VEHICLE CONFIGURATION

The winged booster, illustrated in Figure 4.2-1, is a tri-delta flying wing, consisting of a multi-cell pressure vessel of tapered, intersecting cones. The wing contour is a supercritical Whitcomb airfoil section with the leading edge modified to improve supersonic and hypersonic performance with essentially no reduction in subsonic performance. The outer panels of the wing, and vent system lines in the wing leading edge, provide the gaseous ullage space for the LH_2 fuel located in the inner two panels of the wing. LO_2 tanks are located in the wing about the c.g., and extend from the wing root to the inboard end of the wing tip ullage tanks.

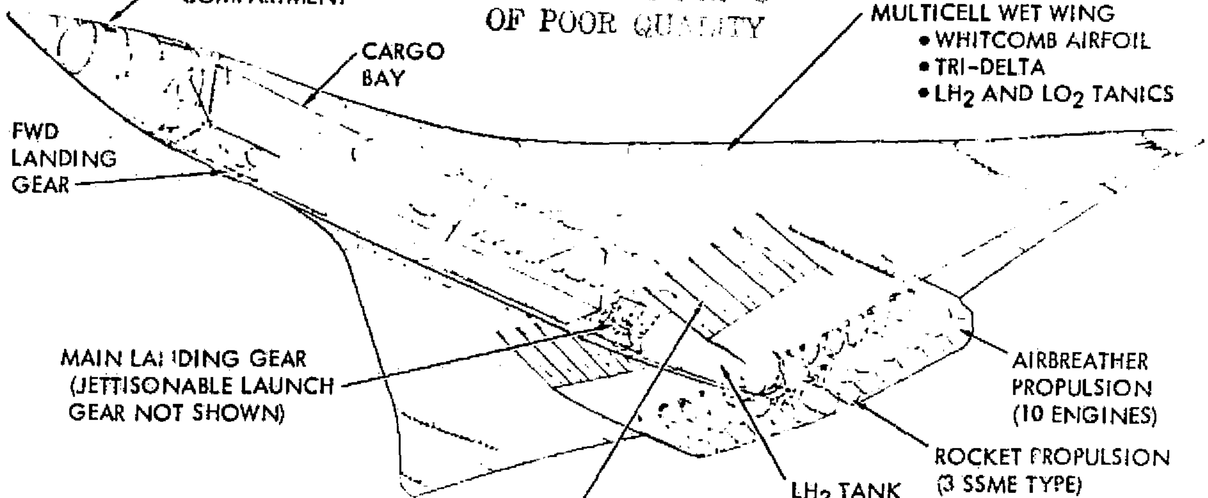

Figure 4.2-1. Winged HLLV - HTO/SSTO Concept

In the aft end of the vehicle, three uprated SSME-type rocket engines (total thrust = 3.2×10^6 lb) are connected to a two-cone LH_2 tank with a double-cone thrust structure. Approximately 50 percent of the volume of the vertical stabilizer is utilized as part of the gaseous ullage volume of the LH_2 tank.

The cargo bay is located forward of the LH_2 tank. Most of the cargo bay side walls are provided by the root-rib bulkhead of the LH_2 wing tank. The cargo bay floor is designed similar to the C5-A military transport aircraft; this permits the use of MATS and Airlog cargo loading and retention systems. The top of the cargo bay is a moldline extension of the wing upper contours, wherein the frame inner caps are arched to resist pressure at minimum weight. The forward end of the cargo bay has a circular seal/docking provision to the forebody. Cargo is deployed in orbit by swinging the forebody to 90 or more degrees about a vertical axis at the side of the seal, and transferring cargo from the bay on telescoping rails. Recapture and reloading of the cargo in space is the reverse of the procedure.

The forebody is an RM-10 ogive of revolution with an aft dome closure. The ogive is divided horizontally into two levels. The upper level provides seating for crew and passengers, as well as the flight deck. The lower compartment contains electronic, life support, power (fuel cell), and other subsystems including spare life support and emergency recovery equipment.

Ten high-bypass, supersonic-turbofan/airturbo exchanger/ramjet engines with a combined thrust of 1.4×10^6 lb are mounted under the wing. The inlets are protected by retractable ramps that close the inlets and fair the bottom surface into a smooth, continuous surface suitable for Sanger skip glider or high angle-of-attack ballistic reentry.

The inboard profile of the winged booster, Figure 4.2-2, illustrates the details of body construction, crew compartment, cargo bay length, LH_2 fuel tank configuration, and location of the rocket engines at rear of fuselage. The hinging and rotation of the nose section for loading and unloading the

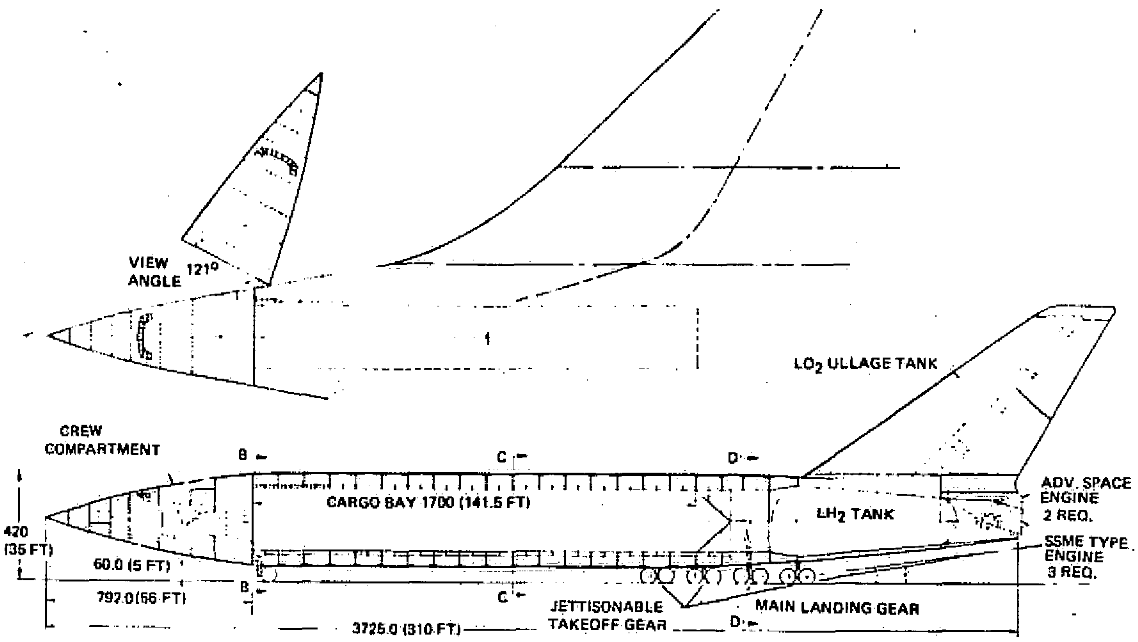

Figure 4.2-2. Winged HLLV - HTO/SSTO Design Details

payloads are illustrated with indication of view angle from rear of nose section during these operations. The multiple landing gear concept shows the positioning of the nose gear bogie, the jettisonable takeoff gear, and the position of the main landing gear for landing with the light, unloaded vehicle. Figure 4.2-3 presents a size comparison of the winged booster and the USAF C5-A galaxy.

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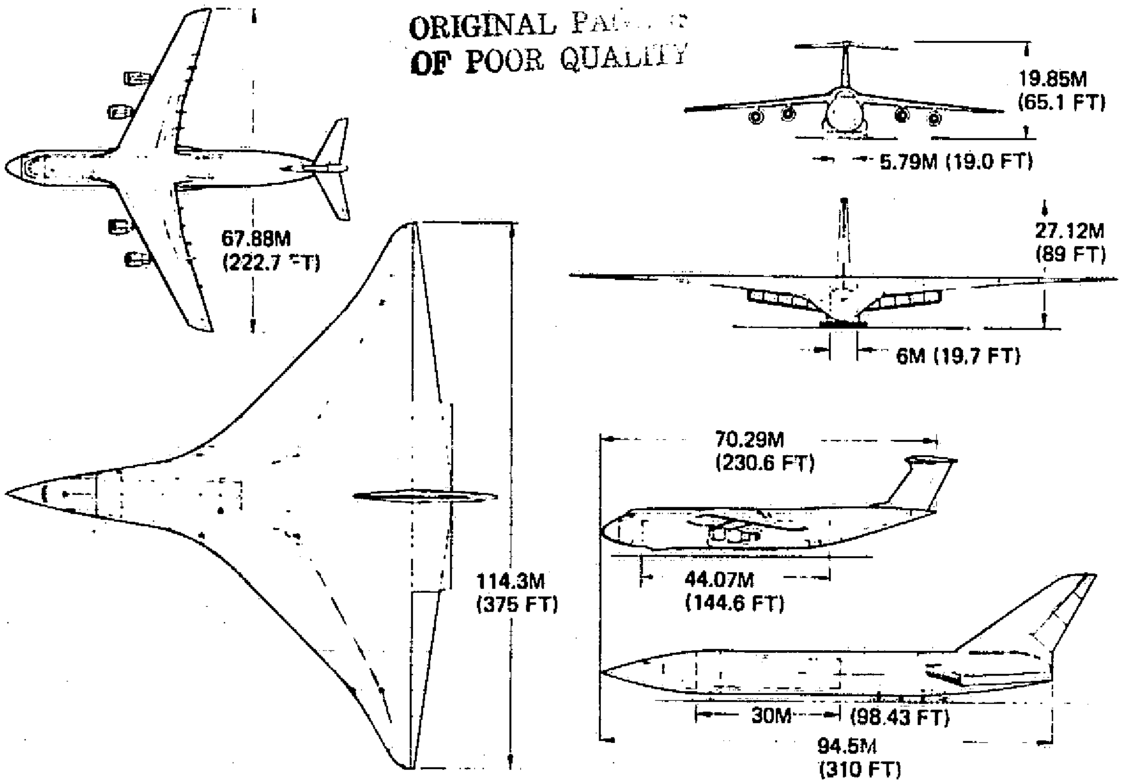

Figure 4.2-3. Winged HLLV - HTO/SSTO C-5A Galaxy

The hydrogen-fueled multi-cycle airbreathing engine concept is depicted in Figure 4.2-4. The core engine is started like a normal turbojet, at which time there is no hydrogen flow through the outer after-fan turbine. The fan is driven by hot gas from the core engine. When airflow is adequate, partial afterburning is initiated and regenerative flow of hydrogen started. Hot H_2 from the regenerative heating is now run through the fan outer turbine, increasing fan speed. H_2 from the outer turbine exhaust provides full afterburner combustion and maximum takeoff thrust. (Taxiing and takeoff preparation is done on core engine only with afterburner not operating.)

When takeoff thrust is no longer required, the augmentor fuel flow is turned off and the engines operate like a normal high-bypass turbofan engine, thus providing economical cruise.

To provide thrust for acceleration, the afterburner is ignited and regenerative flow started. The fan speed is increased by hot regenerative H_2 flow, with increased mass flow and afterburning.

At about Mach 3, sufficient thrust is achievable from the afterburner operating in a ramjet mode. The core engine inlet vanes are rotated to close off core engine inlet to protect the compressor from high inlet temperatures. The fan continues to rotate, cooled by the regenerative H_2 flow through the fan inlet guide vanes. Sufficient thrust is produced to accelerate to the

Mach 6 point, where rocket engines are started. When rocket engines start, fuel flow to the airbreathing engines is terminated and the inlet ramp moves to the fully closed position.

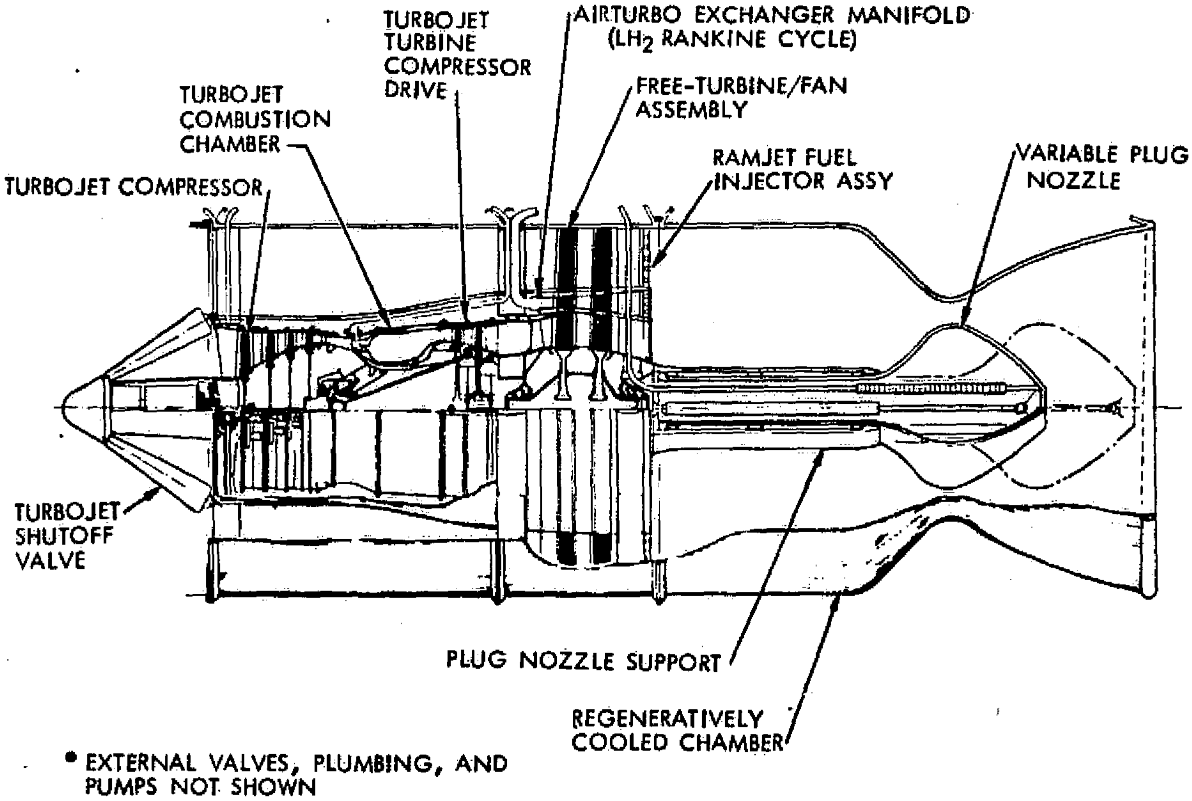

Figure 4.2-4. Multi-Cycle Airbreathing Engine, Turbofan/Air-Turbo Exchanger/Ramjet

The air induction system utilizes a two-dimensional ramp inlet, as shown in Figure 4.2-5, of the mixed-compression type, with five movable ramps which

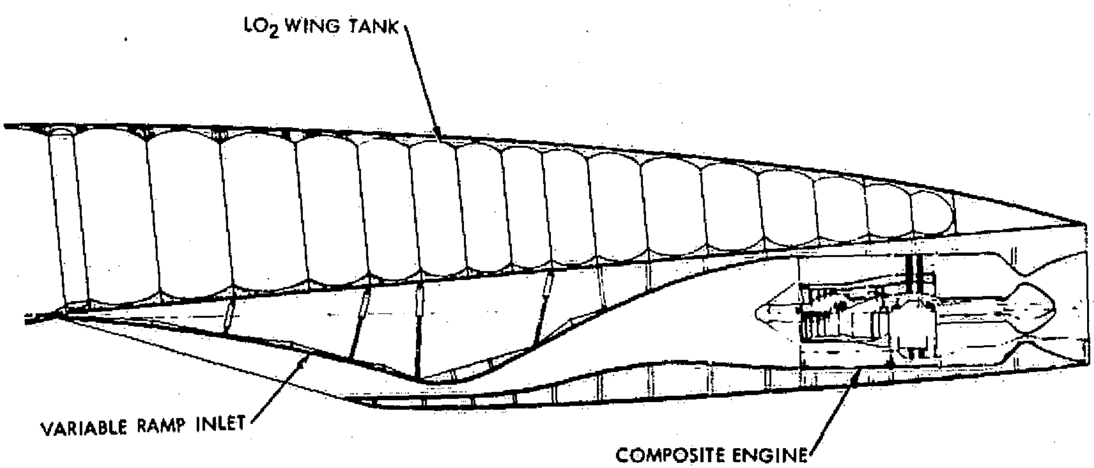

Figure 4.2-5. Inlet and Engine Installation

permit the high contraction ratio required at high supersonic speeds, and which retract to provide the large area required at takeoff and low speed. The ramps also move to completely cover the inlet during rocket engine boost and reentry.

Cooling of the inlet surfaces will be accomplished as required by LH_2 flow through double-wall passages prior to burning in the engine.

4.2.2 LAUNCH WINDOW

Unlike the ballistic HLLV, the winged vehicle is capable of cruise to the equatorial plane prior to injection into LEO. Therefore, there are 12 orbital rendezvous opportunities to a particular orbit with essentially a continuous launch window.

4.2.3 FLIGHT PROFILE AND PERFORMANCE ANALYSIS

The winged booster trajectory is presented in Figure 4.2-6. Takeoff is accomplished under high-bypass turbofan/air-turboexchanger power, with the ramjet acting as a supercharged afterburner. After clearing the runway, the launch landing gear truck is jettisoned and recovered by parachute. The vehicle then proceeds to climb to optimum cruise altitude and Mach number under turbofan power only. At cruise altitude, excess airbreathing engines are shut down to provide economical cruise to the equatorial plane. A large radius turn is then executed into the equatorial plane, the idel airbreathing engines reignited, and a subsonic climb to a suitable altitude is accomplished under turbofan/air-turboexchanger power. A pitch-over into a constant energy, shallow-angle dive is then executed to accelerate through the transonic region; after which, the

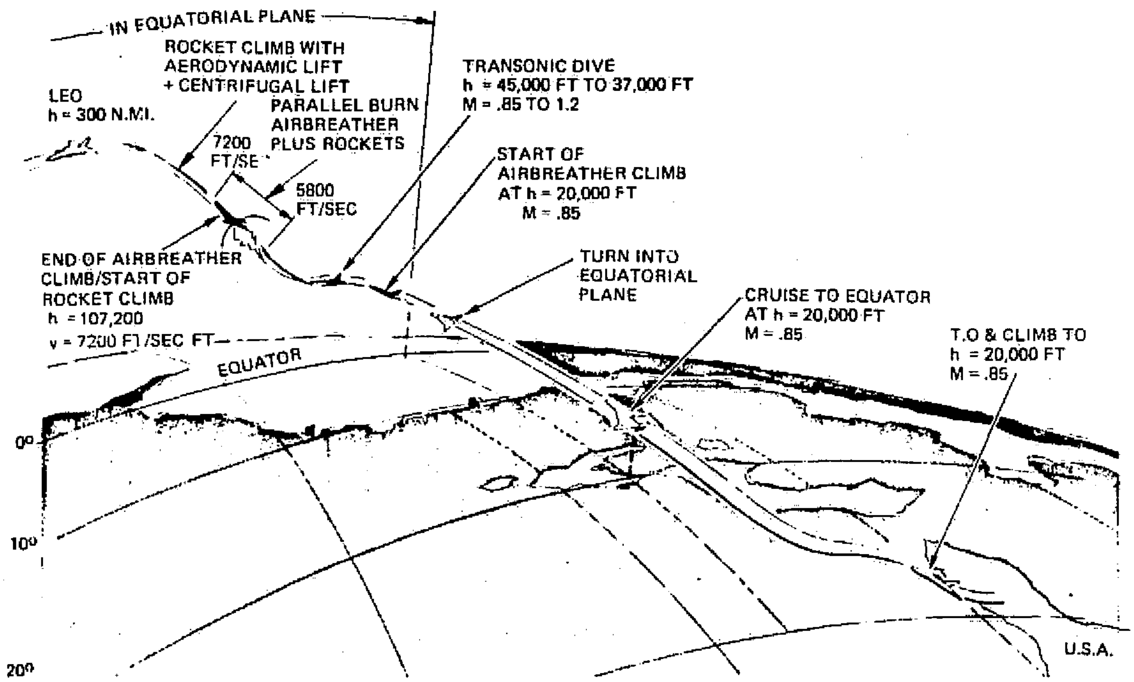

Figure 4.2-6. Winged HLLV - HTO/SSTO Trajectory

vehicle will pitch up into a supersonic climb attitude--still under turbofan/air-turboexchanger power. At approximately Mach 3 and 85,000-ft altitude, the airbreathing engines transition to the ramjet mode and the turbojet shutoff vanes are closed to limit turbine machinery temperatures. The rocket engines are ignited at approximately 100,000 ft and 6200 ft/s, and burn in parallel with the ramjets. The ramjets are shut down and the air induction system closed at Mach 7.2 and 130,000 ft. The vehicle continues ascent to an elliptic equatorial orbit of 91×550 km and the rocket engines are then shut down. A Hohmann transfer into circular orbit is then executed with the auxiliary propulsion system.

For reentry, the auxiliary propulsion system provides the ΔV required for deorbit. A low-flight-path-angle, high-angle-of-attack deceleration maneuver is executed to approximately Mach 6. Partial plane changes are accomplished during this deceleration period. The angle of attack is then reduced to achieve maximum lift/drag for high-velocity glide to subsonic velocity. At approximately Mach 0.85, the inlets are opened and sufficient airbreathing engines are ignited for powered flight to the launch site and vehicle landing.

5.0 LEO-TO-GEO TRANSPORTATION SYSTEMS

5.0 LEO-TO-GEO TRANSPORTATION SYSTEMS

Independent of SPS assembly location, there is a significant demand for OTV transportation from LEO to GEO due to the magnitude of the SPS program. For the LEO-assembled SPS, hardware flights dominate the early years but in later years, logistics flights become a significant factor. Since propellant to support these OTV flights represents a significant portion of the total HLLV payloads, alternate advanced OTV concepts having high specific impulse appeared to be worthwhile candidates to compare to the conventional chemical OTV concept. The OTV concepts chosen for comparison in this study are summarized in Table 5.0-1.

Table 5.0-1 OTV Concepts Evaluated

Cargo	Personnel/Priority Cargo
• Chemical - common stage • Chemical - integrated payload • Electric Self-propelled Dedicated	• Chemical common stage • Nuclear gas core reactor

5.1 CHEMICAL OTV CONFIGURATIONS

Two sizes of common-stage OTV were evaluated; one having a payload capability of ~400,000 kg for use with the ballistic HLLV, and the other having a payload capability of 91,000 kg for use with the winged HLLV. In addition, two options for chemical OTV operation were considered, ground or orbit based. The ground-based option is applicable to the winged HLLV configuration because of its inherent down-payload capability.

5.1.1 CHEMICAL COMMON-STAGE OTV - BALLISTIC HLLV

Several chemical OTV options exist: single-stage, stage-and-a-half, and two-stage, along with several propellant combinations. Among all these options, the common-stage chemical (two stages having the same propellant capacity), utilizing LOX/LH_2 propellant, represents the best compromise between cost and mass required in low earth orbit.

Of the two options for OTV operation, ground- or orbit-based, the latter was chosen for this application. Each option has its own particular set of problems and attractive features but, again, due to the large traffic rate resulting from the SPS program size and the down-payload limitations of the ballistic HLLV, the savings in transportation cost for an orbit-based OTV would more than offset the cost of a depot to support OTV operations.

Configuration

The common-stage chemical configuration, shown in Figure 5.1-1, was taken from the Boeing Aerospace Company *Systems Concepts for SPS-Derived Heavy-Lift Launch Vehicles Study* (NAS9-14710), dated September 1976. The propellant tankage for both stages is the same, but the propulsion system for the first stage has twice as many engines as the second. This allows the thrust/weight for both stages to be kept at about 0.15 g. The engines have a specific impulse of 470 at a mixture ratio of 6:1.

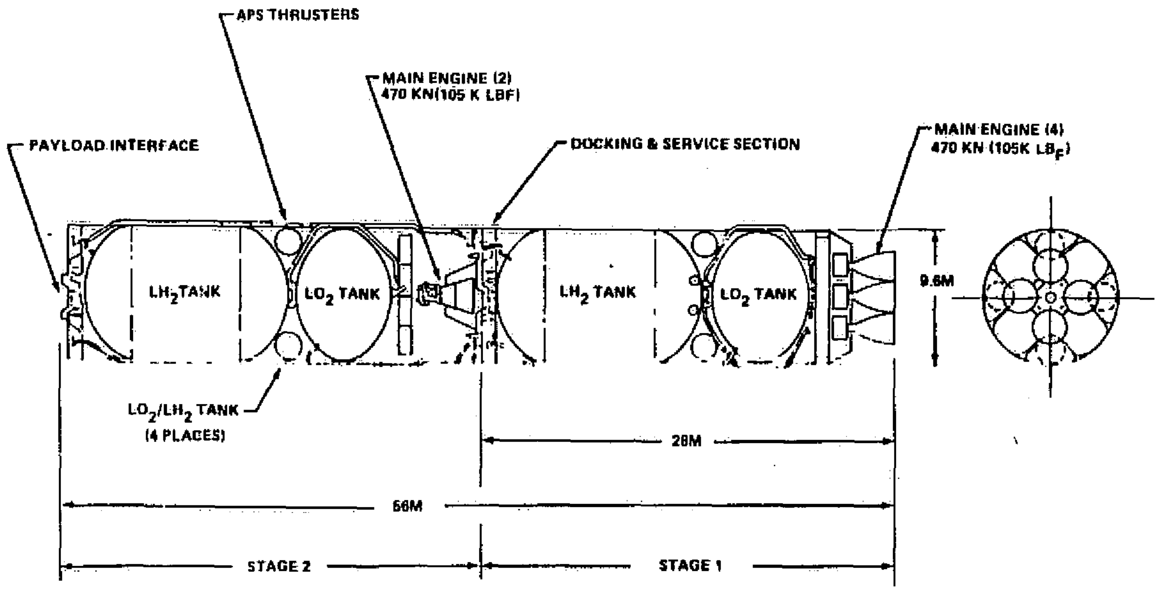

Figure 5.1-1. Common-Stage Chemical OTV Configuration

Mass Properties

Weight data for the common-stage chemical system is given in Table 5.1-1.

Mission Analysis

The mission profile for a high-thrust system consists mainly of a Hohmann transfer. Both the chemical and GCR OTV's are high-thrust systems. Figure 5.1-2 shows the mission profile with the significant events noted. The total flight time is approximately 11 hours; this time, along with 19 hours allowed for the rendezvous, dock and payload transfer, produced a total mission time of 30 hours.

Performance

The OTV performance is given in Figure 5.1-3, where payload is shown as a function of departure inclination. The performance was computed based on a departure/return orbital altitude of 500 km.

Table 5.1-1. Common-Stage OTV Weight Summary

STAGE ELEMENT	OTV WEIGHT (kg) STAGE I	STAGE II
STRUCTURES AND MECHANISMS	9660	10200
MAIN PROPULSION	6580	4800
AUXILIARY PROPULSION	920	5670
AVIONICS	1160	1160
ELECTRIC POWER	1320	1520
THERMAL CONTROL	2300	2690
GROWTH (15%)	3460	4120
DRY WEIGHT	25400	30160
OTHER PROPELLANTS AND FLUIDS	630	3030
TOTAL INERT WEIGHT	26030	33190
MAINSTAGE PROPELLANTS		
LOX	355840	355940
LH_2	59190	51190
STAGE WEIGHT	441064	440220

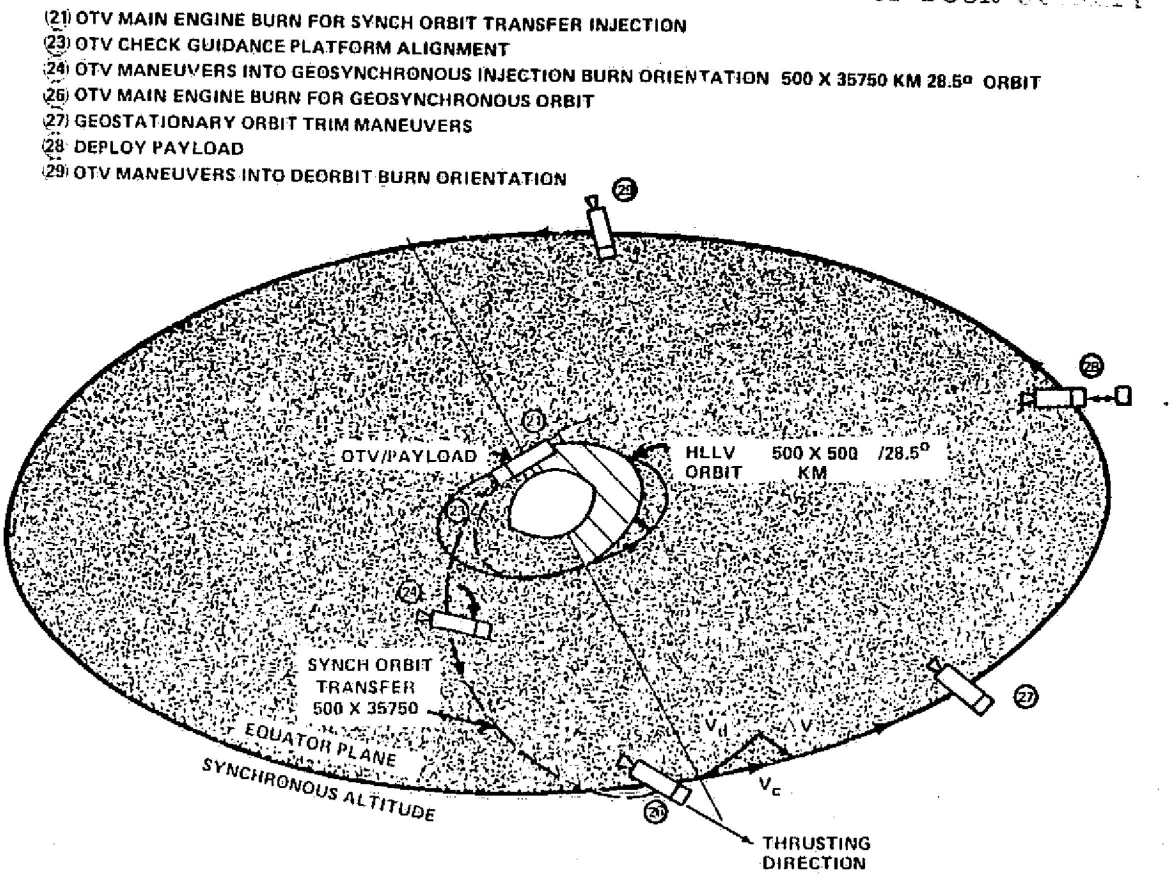

Figure 5.1-2. High-Thrust Trajectory Profile to Geosynchronous Orbit

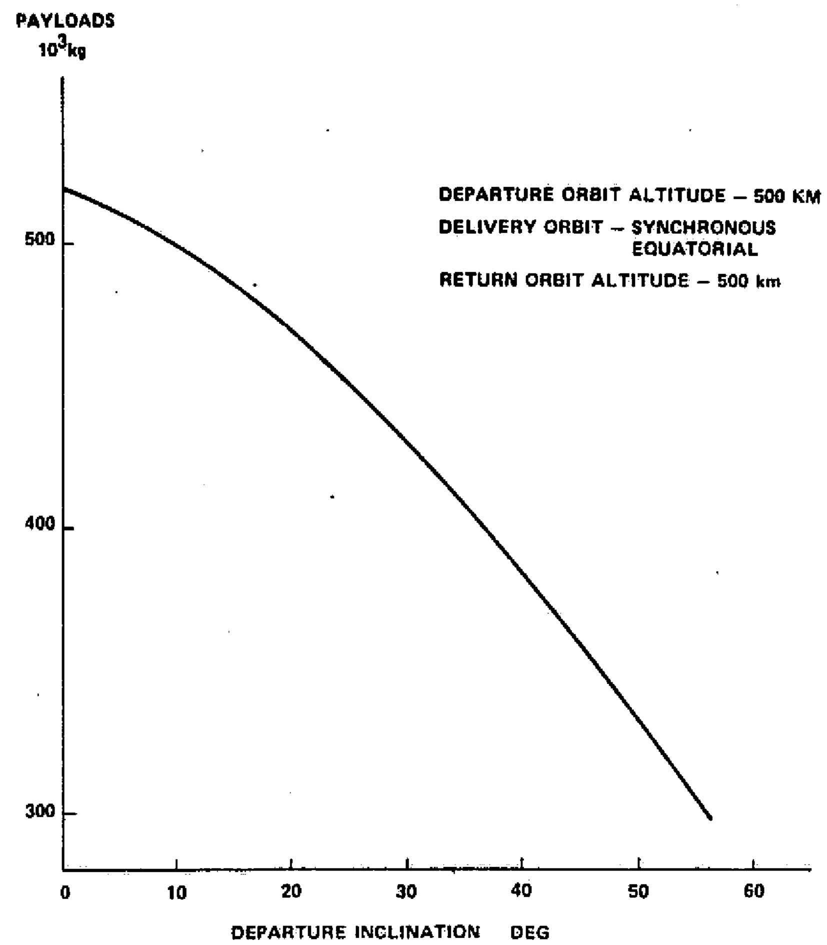

Figure 5.1-3. Common-Stage Chemical OTV Performance Variation with Departure Inclination

5.1.2 CHEMICAL COMMON-STAGE OTV - WINGED HLLV

This common-stage configuration is a scaled version of the ballistic HLLV OTV configuration described above. The payload capability was reduced to meet the payload capability of the winged HLLV. This approach requires three winged-HLLV flights to deliver a 91,000-kg payload to GEO, Figure 5.1-4. The first and second stages and payload module would be assembled on orbit. Following the LEO-GEO mission, the spent OTV stages would be recovered in LEO by subsequent winged-HLLV vehicles and returned to earth for refueling, refurbishment, and reuse.

Configuration

The OTV configuration is given in Figure 5.1-5. The overall length, diameter, tank structures, and docking mechanisms are identical. The only significant difference in both stages are the number of engines--four for the first stage, and two for the second stage.

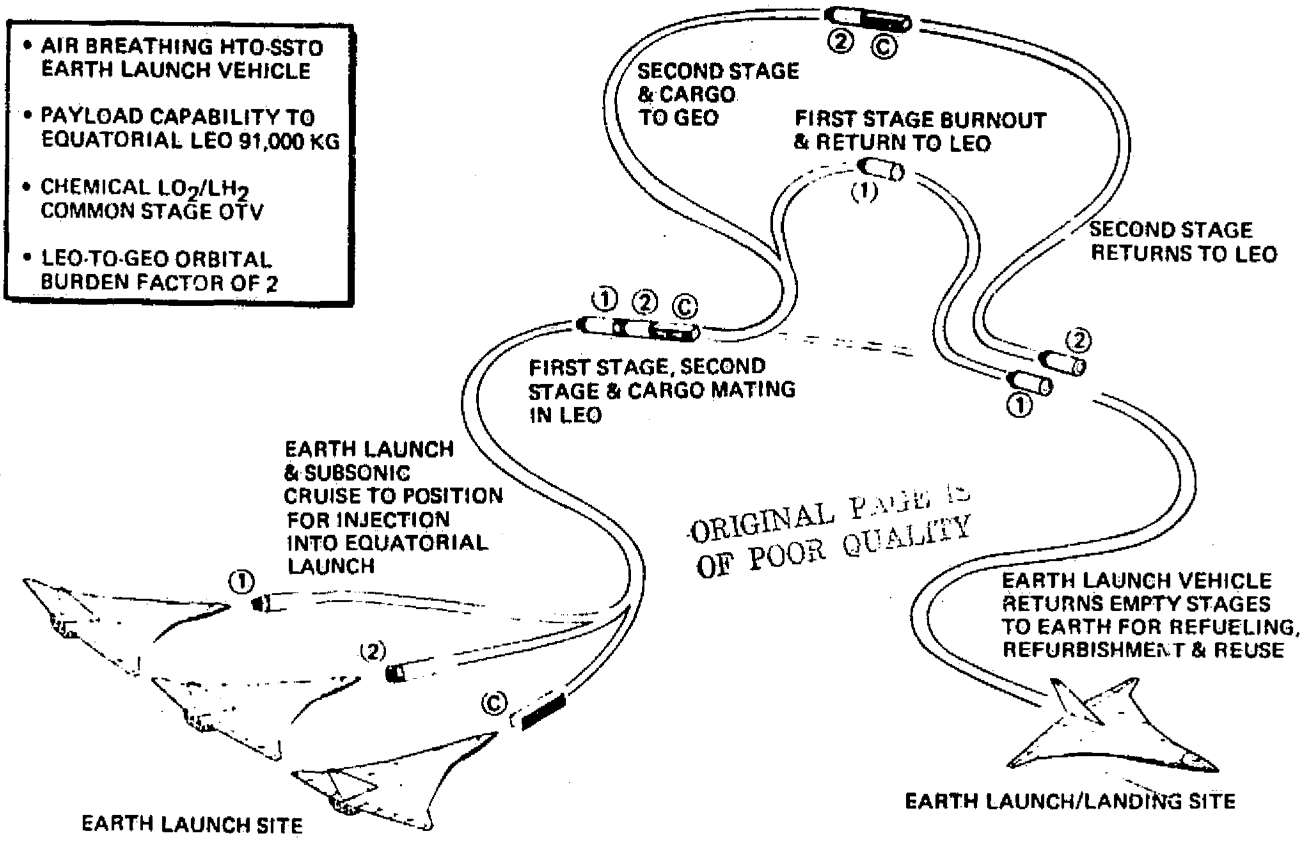

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Figure 5.1-4. Cargo Delivery Concept for GEO Construction of SPS, Large Common-Stage OTV

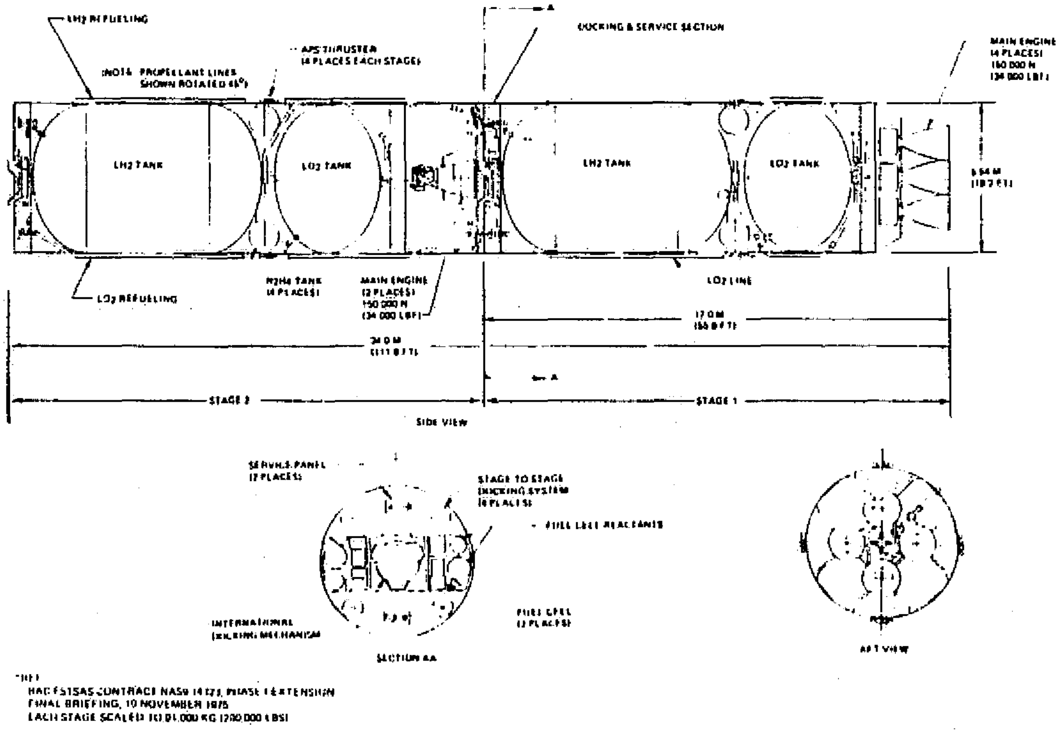

Figure 5.1-5. Common-Stage LO_2/LH_2 OTV Concept

5.1.3 CHEMICAL OTV WITH INTEGRATED PAYLOAD - WINGED HLLV

An alternate chemical OTV approach evaluated is an integrated OTV stage and payload within a single launch vehicle, Figure 5.1-6. This approach precludes the requirement for on-orbit assembly, and reduces the payload packaging requirement. In addition, OTV stage size may be optimized to improve the orbital burden factor over that of the common-stage OTV (i.e., from 2.0 to 1.8). For this case, the maximum payload to GEO is approximately 32,500 kg (71,500 lb) per winged HLLV flight. The integrated OTV/payload for a ballistic HLLV would not yield as great a reduction in orbital burden factor over the common-stage approach.

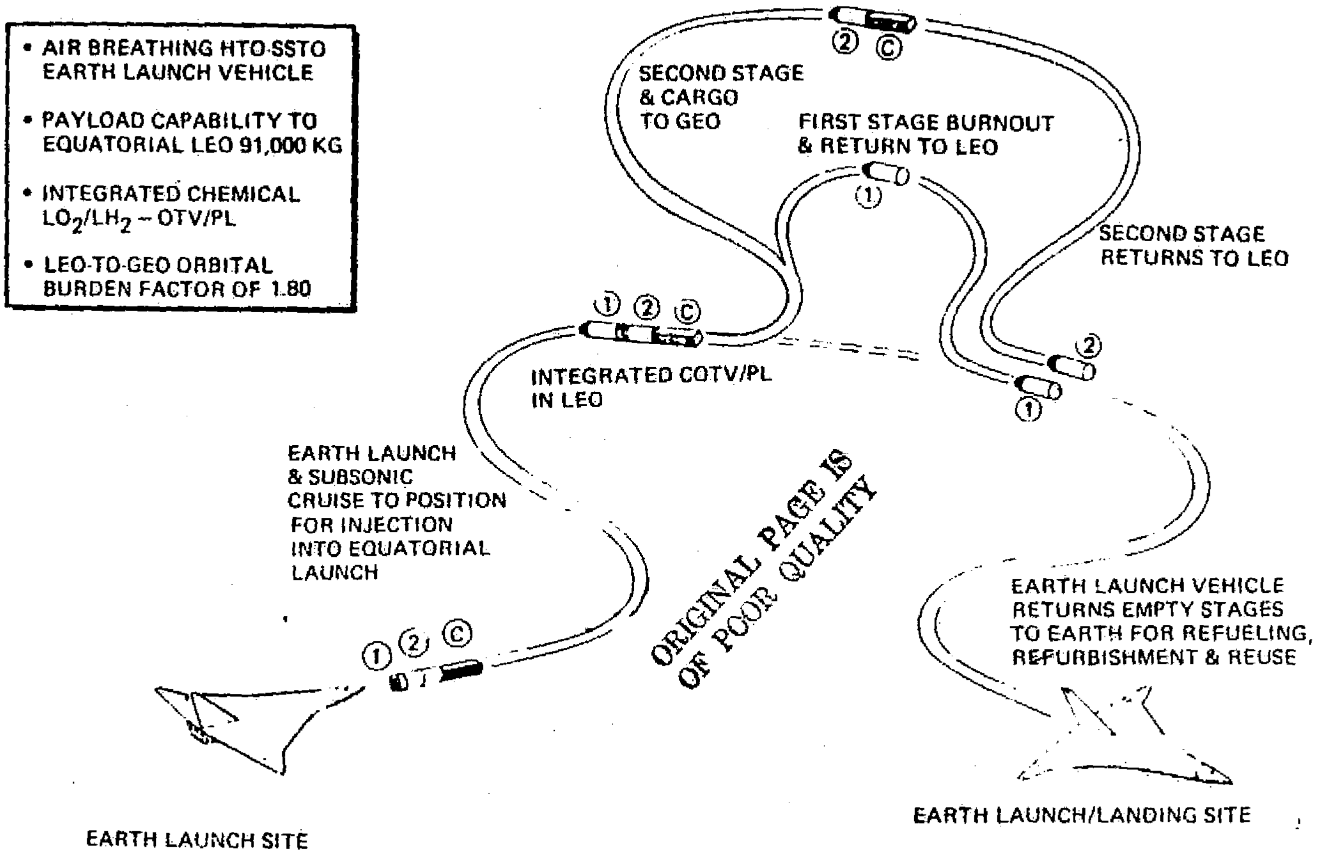

Figure 5.1-6. Cargo Delivery Concept for GEO Construction of SPS, Integrated COTV/PL - Both Stages Returnable

<u>Mass Properties</u>

The mass properties for this configuration is presented in Table 5.1-2.

Table 5.1-2. Integrated COTV/PL Mass Properties

Subsystem	Weight - kg (lb) First Stage	Weight - kg (lb) Second Stage
Structure and mechanisms	663	388
Main propulsion	907	455
Auxiliary propulsion	121	61
Avionics	182	182
Electrical power	154	154
Thermal control	182	91
Weight growth (10%)	221	133
Total dry weight	2,430 (5,860)	1,464 (3,227)
Unusable propellant	209	104
Unusable/reserve APS	9	5
Fuel cell reactants	55	55
Total burnout weight	2,703 (5,960)	1,628 (3,587)
Boiloff/stop-start	138	68
Nominal APS propellant	52	27
Usable propellant*	35,674 (78,645)	18,286 (40,313)
Total start-burn weight	38,567 (85,025)	20,009 (44,110)
Payload to GEO		32,424 (71,486)

*Includes 2% ΔV reserve.

5.2 NUCLEAR OTV

Gas core nuclear rocket engines have been the subject of analytical studies since the mid-1960's. In particular, the nuclear gas core light bulb engine, which has been under study by the United Technologies Research Center appears to be feasible by the 1995-2000 time period. The principal advantage of nuclear systems over other systems, such as chemical and electric, is high I_{sp} (1000 5000 s) and high thrust (222,000 to 2,220,000 N). The specific advantage of the light bulb gas core engine over other gas core reactor propulsion system concepts is that in the light bulb concept, the gaseous nuclear fuel (U-233) does not come in contact with the propellant (LH_2), nor is any of the gaseous nuclear fuel expelled from the engine (closed-cycle). Energy is transferred in the engine by thermal radiation from the gaseous fissioning plasma core, contained by a neon vortex, to the hydrogen propellant which has been seeded with tungsten particles to increase its opacity to radiant energy. The vortex and propellant regions are separated by an internally cooled fused-silica transparent wall (the "light bulb"). The light bulb engine is currently in the technological feasibility phase of development; associated advanced technology analytical studies and hardware testing are being funded by NASA/OAST in support of the gas core reactor program. The characteristics shown in Table 5.2-1 are for a range of light bulb engine thrust levels, all operating at 500 atmospheres pressure (272,000 N/m^2). The lower numbers are for an OTV with a payload to GEO of approximately 227,000 kg, while the higher numbers are for an OTV with a GEO payload of 900,000 kg. The nuclear gas core light bulb engine could be developed with sufficient (i.e., increased) funding and be operational by 1995.

Table 5.2-1. Nuclear Gas Core Light Bulb Engine Propulsion Characteristics

PROPELLANTS	LH_2 (U-233 NUCLEAR FUEL)
SPECIFIC IMPULSE	2080 – 2425 SEC
THRUST	445,000 – 1780,000 N
ENGINE WEIGHT	42,000 – 91,000 kg
PRESSURES	271,000 N/M^2 (OPERATING) 384,000 N/M^2 (MAXIMUM)
CYCLE	CLOSED – CYCLE (NO U-233 IN EXHAUST GAS)
TECHNOLOGY STATUS	ADVANCED TECHNOLOGY

5.2.1 CONFIGURATION

The nuclear stage, shown in Figure 5.2-1, has a core reactor propulsion unit that produces 890,000 N of thrust at a vacuum specific impulse of 2250 s. The hydrogen fuel is contained in the large spherical tank. The other systems--attitude control, avionics, and mechanical--are situated in the conical transition sections.

5.2.2 MASS PROPERTIES

The weight data for the nuclear OTV is shown in Table 5.2-2. The main propulsion system accounts for more than half of the total stage inert weight.

5.2.3 MISSION ANALYSIS

The mission profile and total flight time is similar to the second stage of the chemical OTV.

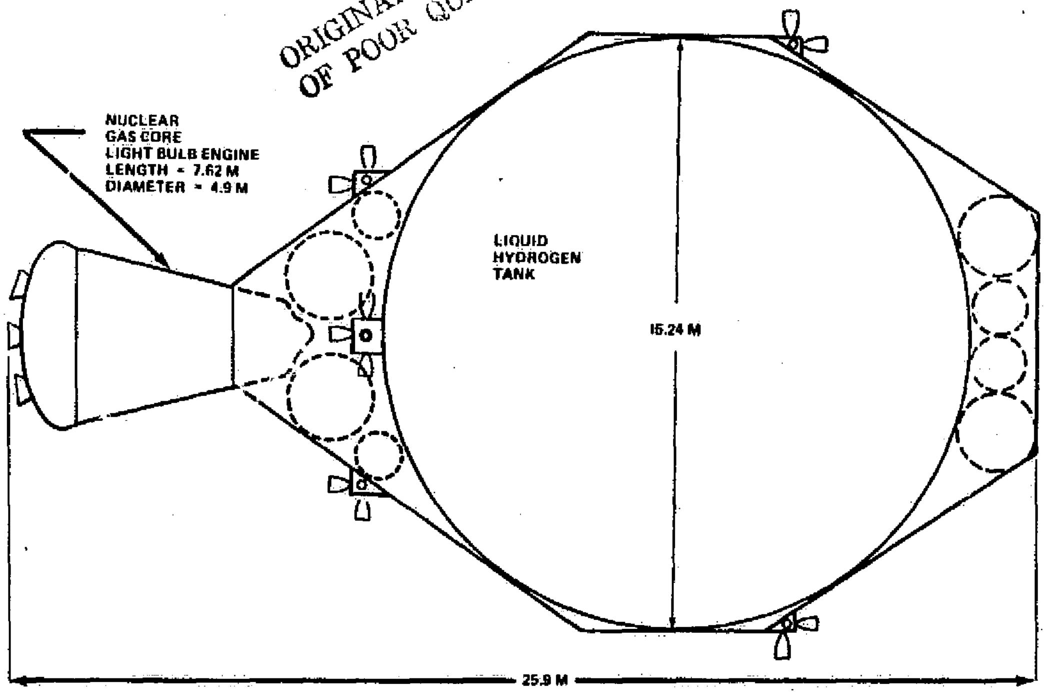

Figure 5.2-1. Nuclear Gas Core Reactor OTV

Table 5.2-2. Nuclear Gas Core Reactor OTV Weight Summary

STAGE ELEMENT	WEIGHT Kg
STRUCTURES AND MECHANISMS	18780
MAIN PROPULSION	56850
AUXILIARY PROPULSION	600
AVIONICS	260
ELECTRIC POWER	480
THERMAL CONTROL	1220
GROWTH (15%)	11730
DRY WEIGHT	89920
OTHER PROPELLANTS AND FLUIDS	2000
TOTAL INERT WEIGHT	91920
MAINSTAGE PROPELLANTS	
LOX	
LH_2	124280
STAGE WEIGHT	206204

5.2.4 PERFORMANCE

The nuclear OTV performance is shown in Figure 5.2-2, where payload is shown as a function of departure inclination. The performance was computed based on a departure/return orbital altitude of 500 km.

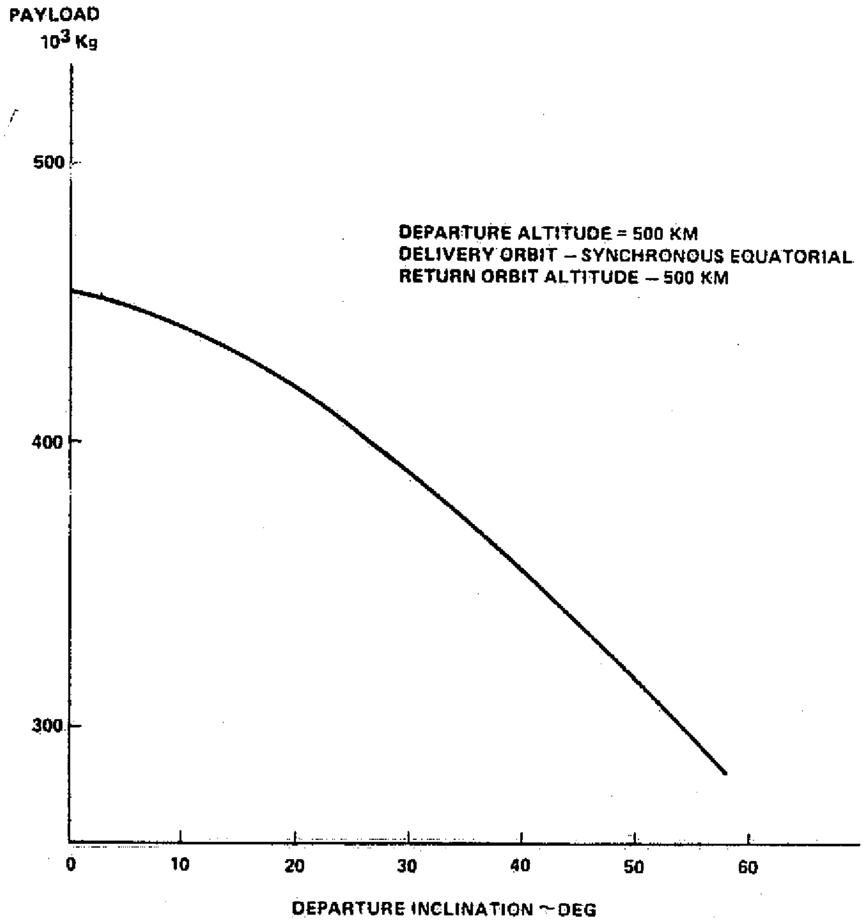

Figure 5.2-2. Nuclear Gas Core Reactor OTV Performance Variation with Departure Inclination

5.3 ELECTRIC ORBITAL TRANSFER VEHICLE

The requirements for low-thrust, high-specific-impulse propulsion systems to transfer satellites or their materials from LEO to GEO are described. The rationale for selection of argon ion bombardment electric thrusters is summarized, and the performance parameters used for these thrusters is defined. Power sources and conditioning and argon propellant storage and distribution concepts are described. Design trades and baseline concepts are presented for electric propulsion systems associated with two SPS construction modes: partial LEO construction, and all-GEO construction.

5.3.1 CONSTRUCTION AND PROPULSION OPTIONS

It was previously shown that a chemical orbital transfer vehicle (OTV) requires a prohibitive propellant mass to place the SPS mass in GEO. This is because of the limited (<490 s) specific impulse (I_{sp}) of foreseeable chemical systems. Therefore, high-I_{sp} thrusters present an attractive alternate for the cargo OTV.

The partial LEO construction mode was first considered for the SPS. This mode necessitates a low-thrust-to-mass ratio to avoid structure design penalties; a maximum ratio of 9.8×10^{-3} m/s^2 (10^{-4}g) is required. Although such ratios can be achieved with chemical systems, the trip time imposes the requirement for storable propellants (I_{sp} ~338 s), thus further increasing the propellant mass requirements.

The all-GEP construction mode avoids load limitations on the packaged SPS materials, but the high I_{sp} requirement continues to demand electric propulsion for the OTV. The OTV solar power source (photovoltaic--as in this study--or thermal) is characterized by large, lightweight structures comparable to the SPS. Therefore, a low thrust to total mass in LEO is again required, even if it were practical to provide high thrust from a sufficiently large thruster array and solar array or collector.

A lower limit on the thrust-to-mass ratio is established by the trip time to GEO and back to LEO. Radiation degradation of electric OTV solar cells is traded against thrust and trip time in Section 5.3.2. An operational limit is set by the time between successive departures of satellites from LEO. Two OTV sets can support a construction rate of four per year if the round-trip time (plus turnaround operations) does not exceed 182 days. A third OTV set would be needed to support the five-per-year satellite construction rate in the 2021 to 2025 time period.

5.3.2 ELECTRIC PROPULSION TECHNOLOGY OPTIONS

The major technology options for the electric OTV propulsion subsystem concern the thruster type, size, and design operating point; the power interfaces between the thrusters and the solar array or other primary source; and the propellant type, storage, and distribution.

Thruster Type

Thruster types considered for this application were ion bombardment, magnetoplasmadynamic (MPD), and resistojet. Other types, such as RF excitation, were rejected *a priori* because of development risk and lack of evidence of performance superior to the types first mentioned.

Resistojet thrusters were discarded because their low I_{sp} (<1200 s) offers insufficient propellant mass savings compared to chemical propulsion.

MPD thrusters were initially considered on the basis of reported I_{sp} values up to 10,000 s.[1] To establish the validity of such data, Rockwell supported an independent investigation by Dr. G. L. Cann of Technion, Inc. (State of the Art of Electromagnetic and Electrothermal Propulsion, Technion, Inc., Report No. 07-040, July 18, 1977). He established that high I_{sp} values were measured in small vacuum chambers which allowed exhaust propellant to be recirculated through the thruster; this appeared to reduce the propellant flow rate and proportionately increase I_{sp}. The state-of-the-art I_{sp} is actually in the range of 2000 to 2500 s, with 4000 s the realistic growth potential. For this reason, and because MPD thruster development has been largely abandoned except for long-range research at Princeton University, this type was dropped from consideration.

The surviving candidate, for which a current development program has established reliable performance data, is the ion bombardment thruster.

Power Conditioning

Conventional power conditioners for ion bombardment thrusters regulate all supplies, serving as an interface between the power source (solar array) and the thrusters. Various so-called direct-drive concepts have been proposed in which some of the thruster supplies are obtained directly from the solar array. This approach reduces power conditioner mass, power loss, and cost, and improves propulsion system reliability. Possible disadvantages are associated with solar array power losses and temperature variations that result in voltage changes.

The power conditioners of the SPS propulsion system process only the low-voltage fixed power (278 W input per thruster). The other supplies are taken directly from solar arrays. The beam power is obtained from the main SPS or OTV solar array. To avoid significant power loss from plasma discharge, the array voltage is maintained at 2000 V; this is stepped up to the beam voltage by dc-dc converters before collection by the main solar array power distribution lines. These lines can be well insulated against discharge losses without significant mass penalty. Solar eclipse produces solar cell temperature, efficiency, and output voltage variations which cause acceptable transients in the beam voltage during the first few minutes after each eclipse.

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[1] *Large Payload Earth-Orbit Transportation with Electric Propulsion,* J. W. Stearns, Tech Memo 33-993, September 15, 1976.

The accelerator and discharge power sources are small solar arrays near the thrusters. This location reduces cabling mass at the low voltage involved; plasma discharge is negligible. Because only 50 kW per thruster is generated, thermally induced voltage transients can be regulated by voltage limiters. An auxiliary power unit (APU), charged by the discharge supply solar array, furnishes 278 W at 90-percent efficiency to the thruster low-voltage supplies. The power sources and conditioning are illustrated in Figure 5.3-1.

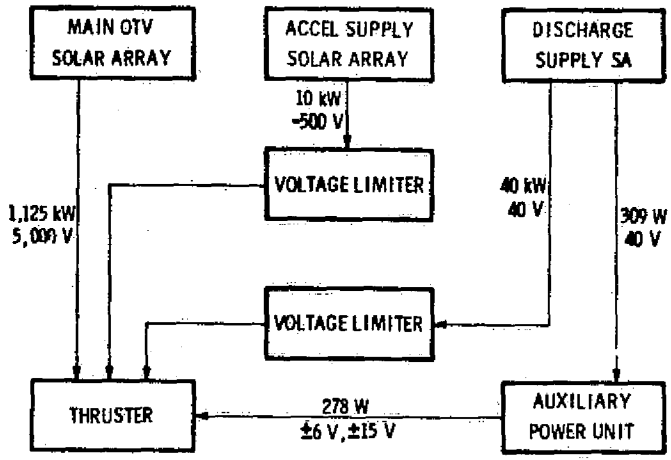

Figure 5.3-1. Power Source and Conditioning

<u>Propellant Selection</u>

The ion thruster propellant selection criteria are availability, storability, absence of serious environmental impacts, cost, demonstrated performance, and technical suitability. Availability becomes a major issue when it is recognized that more than 10^6 kg of propellant is required for one satellite. Technical factors are as follows.

- *High specific impulse* - At a given beam voltage, $I_{sp} \sim 1/\sqrt{m_i}$, where m_i is the ion mass.
- *High thrust* - At a given beam voltage and current, $T \sim m_i$.
- *Low vaporization temperature* - Allows instantaneous thruster restart after solar eclipses without power storage for preheating.
- *Low first-ionization potential* - Limits thruster discharge loss and minimizes the efficiency loss due to neutral atoms.
- *High second-ionization potential* - Minimizes the efficiency loss due to multiple ions.

Obviously, the first two factors are mutually contradictory and are best compromised by an ion of medium mass.

The propellants for which ion bombardment thruster experimental data exist are evaluated against the above criteria in Table 5.3-1. The selection of argon is self-evident.

Table 5.3-1. Ion Propellant Selection Criteria

PROPELLANT	AVAILA-BILITY	STORABILITY	ENVIRON-MENTAL FACTORS	COST ($/KG)	THRUSTER TECHNOLOGY STATUS	ATOMIC WEIGHT	VAPORI-ZATION TEMP. (K)	IONIZATION POTENTIALS (V) 1	IONIZATION POTENTIALS (V) 2
ARGON	HIGH (0.9% OF AIR)	CRYOGENIC	INERT	0.50	GROUND TESTS	39.9	97	15.76	27.62
CESIUM	PROBABLY INADEQUATE	SOLID	EXTREMELY REACTIVE	300	LABORATORY DEVELOP.	132.9	951	3.89	25.1
XENON	VERY SCARCE	CRYOGENIC	INERT	1000	LABORATORY DEVELOP.	131.3	167	12.13	21.2
MERCURY	MARGINAL	LIQUID	TOXIC	55	SPACE FLT	200.6	530	10.43	19.13

Argon Ion Thruster Performance

The argon thruster design and performance characteristics used in this study are based on work conducted at NASA Lewis Research Center.[1] Discussions with D. C. Byers and V. K. Rawlin are also gratefully acknowledged.

The thruster aperture diameter (D) was chosen as 100 cm. Experience with the development of 8- and 30-cm thrusters, now at an advanced stage, suggests that the performance of 100-cm thrusters can be analytically predicted with only minor deviations.[1] The cathodes and ion extraction systems require major modifications. Multiple cathodes are employed to improve lifetime, reliability, and performance. It is assumed that more resistant cathodes can be constructed with lifetimes comparable to the OTV (30 years). However, the grid sets will have to be refurbished periodically because of positive ion bombardment. The current replacement concept uses a remotely operated crane, such is found in some modern warehouses, that rides back and forth on a track. The track, in turn, rides on a second track perpendicular to the first, so as to enable grid set exchanges anywhere in the propulsion module matrix. The grid sets are removed by a twist and a pull; analogous to the removal of a bayonet-base light bulb. Grid insertion is accomplished by a push and a twist in the opposite direction. The crane is capable of storing, transporting, and exchanging a complete matrix of grid sets. The grid sets, complete with guides and locks, have an estimated mass of about 5 kg each.

The concept of a dished grid, which proved successful for the 30-cm mercury thruster, appears feasible for 100-cm argon thrusters. Dished grids, which enable closer-spaced accelerator grids, effectively result in greater thrust density but impose a limit on the specific impulse.

[1]Byers, D. C. and Rawlin, V. K., *Electron Bombardment Propulsion System Characteristics for Large Space Systems,* NASA-Lewis Research Center, AIAA 76-1039.

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Space Division

It was assumed in Reference 1 that the 100-cm argon thruster would have ionization losses of about 200 eV/ion with utilization efficiencies of 0.8 to 0.9. Subsequently, non-optimal tests using argon in a 30-cm mercury thruster showed higher losses (300 to 400 eV/ion) and lower efficiencies (0.6). Consequently, in one recent study, the small-hole accelerator grid (SHAG) optics concept was adopted to improve the design.[1] However, it appears at this time that SHAG optics are unnecessary if dished grids are used. In fact, the large, optimized thrusters are expected to have losses as low as 150 eV/ion or lower.[2]

It is believed that the 100-cm thruster is not so large as to give problems in constructing the grids, e.g., out of molybdenum, or in refurbishing on orbit. The maximum working grid temperature was taken to be 973 K, which is far below the projected maximum operational temperature capability of molybdenum which could exceed 1500 K.

Because of the desire to have short trip times, the OTV design was based on a high specific impulse of 13,000 s. The final propulsion module parameters used in this study are summarized in Table 5.3-2.

Table 5.3-2. Propulsion Module Parameters

Thruster module beam current	225 A
Accelerating voltage	5000 V
Propellant utilization eff.	0.82
Total thruster eff.	0.80
Specific impulse	13000 s
Argon ion speed	155.3 km/s
Mean exhaust speed	127.5 km/s
Thruster module electric power	1,203. kW
• Ion jet power	1,153. kW
• Discharge power	40. kW
• Grid set power	10. kW
• Fixed power	0.3 kW
Mechanical power in diverging beam	922.5 kW
Thrust (diverging beam)	14.47 N
Useful thrust (assumed 90%)*	13.02 N
Module mass	120. kg

*Corrected for divergence and double ionization.

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[1] *Solar Power Satellite System Definition Study, Part I, Vol. V*, Boeing Aerospace Company Report D180-20689-5, July 28, 1977 (p. 176).

[2] Private communication between D. H. Robey, Rockwell International, and V. K. Rawlin of NASA-Lewis Research Center (8 February 1978).

Propellant Storage and Distribution

The argon propellant storage, distribution, and reliquefaction subsystems were estimated to be 15 percent of the total propellant system mass. The propellant tank mass was estimated on the basis of current high-performance task design, and represents approximately 5 percent of the system mass. Because of the long feed lines required to service the multi-thruster modules, approximately 6 percent was required for vent/feed lines, isolation and vent valves, and insulation. The need for a reliquefaction system was assumed because of the extensive propellant distribution system. A system sized to handle 5.2 kW represents approximately 4 percent of total system mass.

5.3.3 SELF-POWERED OTV DESIGN

Electric propulsion requirements were determined for LEO-to-GEO transportation of an SPS whose primary structure, rotary joint, and microwave antenna structure are constructed in LEO. Sufficient solar array and concentrator areas are also constructed in LEO to provide power for electric propulsion, including attitude control, during ascent. The remaining solar array and concentrator materials, as well as the microwave transmitter elements, are packaged near the SPS center of gravity, with construction completed in GEO.

The ground rules and performance data listed below were used in this design effort.

1. LEO is 556 km (300 nmi), equatorial.

2. EPM specific mass = 1 or 2 kg/kW, representing 6 or 12 times the maxx of 100-cm argon ion bombardment thrusters (20 kg at 120 kW, with 1.75 N thrust each). (As a point of reference the FSTSA study, Phase II Final Report, Boeing Aerospace Co., Report D180-20242-2, 1977, assumed at EPM specific mass of 1.5 kg/kW.) The EPM's include thrusters, power conditioners, propellant feed lines, cabling, structure, and (if required) gimbals. The specific impulse is 13,000 s. (Note: Higher input power and thrust values were used in later work reported below.)

3. Argon tankage mass, including thermal control for liquid argon at 87 K (-186°C), plumbing, etc., is 15 percent of argon mass.

4. The solar array degradation due to radiation is based on proton irradiation tests of Rockwell GaAlAs solar cells; no annealing was considered here. It is assumed that all degradation occurs at the start of ascent. The actual degradation is approximately

$$P \cdot P_o = 1 - k \ell n (\phi t)$$

where k = constant, ϕt = proton fluence expressed as equivalent fluence of 1 MeV electron. Most of the power loss, therefore, occurs relatively early in the ascent.

5. The power loss due to degradation during transfer to GEO is made up by increasing the SPS size so that the beginning-of-life (BOL) power at GEO is 8.19 GW. This level, after other losses (in addition to radiation damage), gives 5 GW at the rectenna. The SPS mass increase is 2.65 kg/kW, of which 2.36 kg/kW represents the added solar array and 0.29 kg/kW the added basic structure to accommodate the larger array.

6. Ten percent of the EPM's are left with the SPS for station-keeping and attitude control, together with an argon supply equal to 10 percent of the total argon propellant mass required for orbital transfer.

The results are presented in Table 5.3-3 for photovoltaic satellites with geometric concentration ratios of 1:1, 2:1, and 5:1.

Table 5.3-3. Electric OTV Requirements (Integrated OTV)

	CONCENTRATION RATIO		
	1	2	5
SPS Mass if all GEO Construction	42.461	33.940	40.757
Actual SPS Mass	46.898	34.656	41.794
SPS Length Increase (Percentage)	10.5	10.5	0
Electric Propulsion Modules			
Dry Mass	1.685	1.340	1.455
Argon Mass	1.984	1.558	1.713
Thrusters	14,060	11,130	12,140
Total Solar Array Power in LEO	1.870	1.390	1.569
Total Solar Array Power in GEO	1.028	0.764	0.863
Dedicated Solar Array			
BOL Power	0.150	0.109	0.147
EOL Power	0.076	0.047	0.063
Mass	0.492	0.258	0.481
SPS Solar Array Make-Up			
BOL Power (Lost)	0.785	0.591	1.422
Mass	2.566	0.612	0.713
SPS Attitude Control			
Electric Mass	0.066	0.048	0.062
Thrusters	548	396	514
Argon Mass/Year	0.05	0.04	0.05
Chemical Mass Including Propellants	1.4	1.0	1.3
Total Mass in LEO	55.012	39.424	47.456
Orbital Burden Factor	1.30	1.16	1.16

All masses in 10^6 kg.
All powers in GW.

Because of the cost of the electric propulsion subsystem, it is desirable to return this to LEO except for thrusters and other components used for operational stationkeeping and attitude control of the SPS. Return of this hardware by a storable chemical propulsion system was considered. One-way return trip times up to 200 days were considered. The chemical stage and propellant requirements to return spent EOTV's to LEO were calculated for ascent times of 30 to 120 days, and are shown in Figure 5.3-2. Even for ascent times as long as 200 days, the chemical descent OTV mass (including storable propellant) is a significant fraction of the total mass in LEO.

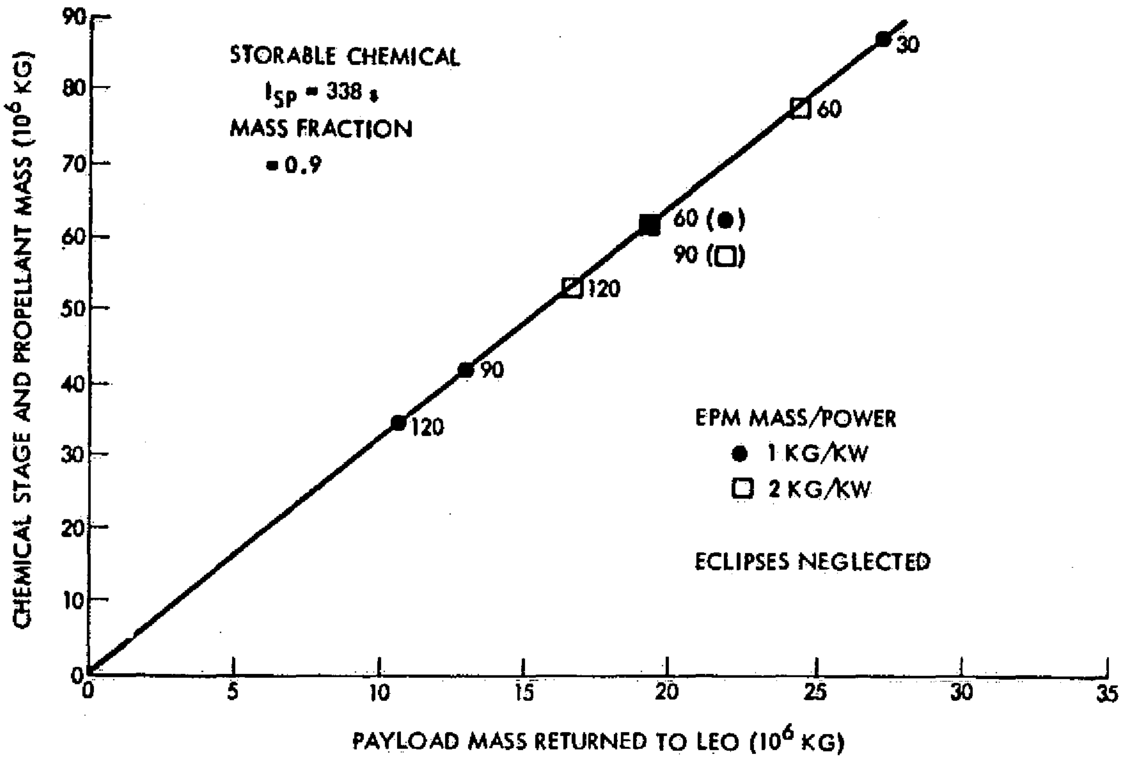

Figure 5.3-2. Chemical Return of Electric OTV's to LEO

The orbital burden factor (mass in LEO/mass in GEO) can be reduced if the EOTV's are returned to LEO by means of electric thrusting. The power is derived from a segment of the SPS solar array deployed in GEO and returned to LEO; this segment is called the dedicated solar array (DSA). The following quantities are plotted vs. ascent time (equal to descent time) in Figure 5.3-3 for a CR = 1 photvoltaic satellite:

- Dedicated solar array power at departure from LEO
- Number of thrusters for ascent
- SPS mass increase to offset solar array damage during ascent
- Mass in LEO, less the SPS mass including the above increase

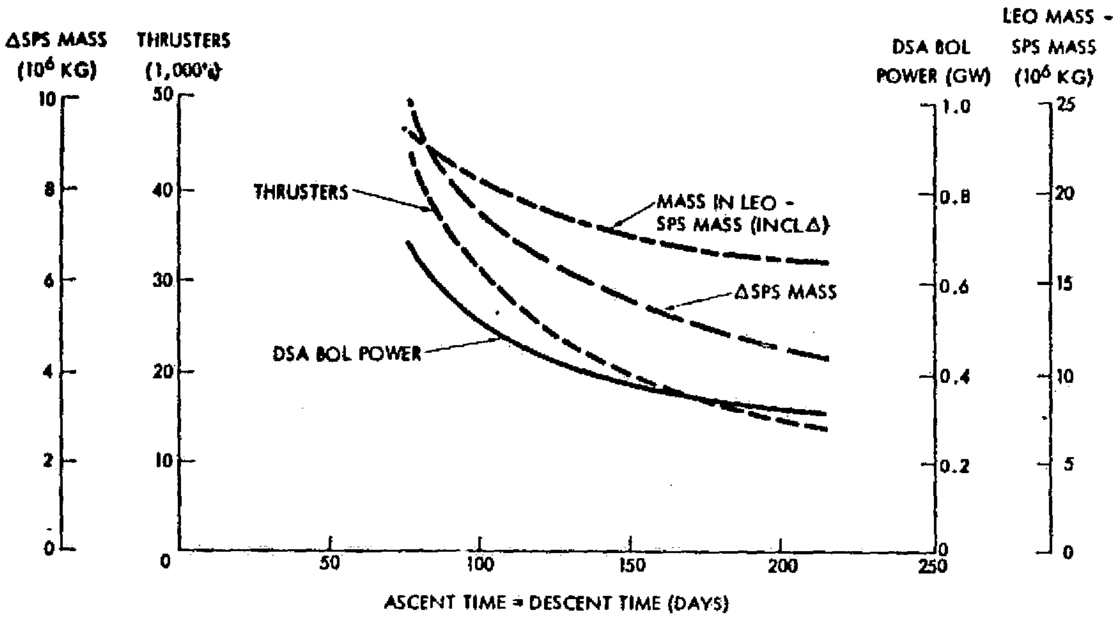

Figure 5.3-3. Electric OTV Requirements Vs. Trip Time (Depart LEO March 21)

The ascent time varies slightly with LEO departure time because of seasonal differences in the fraction of shadowed operation vs. altitude. The variation of trip time is cyclic, with a six-month period and minima at the equinoxes. To keep the trip time constant requires a 3-percent thrust increase for departure near the solstices.

The dedicated solar array approach is clearly superior because of the great savings in the initial mass in LEO.

5.3.4 DEDICATED ELECTRIC OTV DESIGN

The dedicated EOTV is a reusable orbital transfer vehicle designed to transport unassembled SPS payload elements between LEO and GEO. It returns to the LEO facility where it is refurbished for subsequent reuse.

The round trip transfer time was found to be a key parameter in the overall SPS program optimization. The round trip orbital transfer time must remain well below 180 days in order to meet the SPS construction rate of two per year. After establishing time allotments for refurbishment, payload loading, possible reflector cleaning or coating, checkout and refueling, etc., a round trip time of 138 days was selected as a design goal. It was determined that the selected transfer time permitted adequate margin for error and possible HLLV launch delays.

In order to accommodate a shorter trip time without undue penalty in array size and number of thrusters, the results of self-annealing properties of GaAlAs were considered. Studies on radiation recovery of GaAlAs by the Rockwell Space Division and Science Center indicate that cells damaged by electrons and protons with energies up to 10 MeV have shown considerable improvement in power output after annealing at temperatures between 130°C and 240°C. The total recovery time is on the order of one hour at 240°C, 30 hours at 160°C, and 400 hours at 130°C. The fractional recovery is strongly dependent on cell junction depth and construction, and may range from 10 to 80 percent. Curiously, after irradiation and annealing, the cell output may exceed initial specifications by 5 or 10 percent. Because of these available data on annealing, the average solar blanket output in the SPS time frame was taken to be 85 percent of initial BOL specifications.

Another factor entering into OTV sizing was the decision to use SPS baseline solar blanket modules; these modules have an area of 900,000 m^2. The thruster specific impulse, based on NASA-LeRC data, was taken to be 13,000 s, which is in consonance with shorter trip times.

Based on the foregoing assumptions, the sizing of the COTV was accomplished in a straightforward manner.

Solar Blanket and Thruster Power

The available BOL power, P_{SB}, at the output of the solar blankets was determined with the help of the efficiency chain shown in Table 5.3-4.

Table 5.3-4. Electrical Power Efficiency Chain

Power Loss Item	Efficiency		W/m^2
	Individual	Cumulative	
Solar blanket pointing	0.95	0.95	2570.7
Reflector (CR = 2)	0.90	0.855	2313.6
• Reflectivity			
• Degradation			
GaAlAs solar cell eff.	0.176	0.15048	407.2
Solar blanket design	0.9006	0.1354	366.4
• Eff. area			
• Mismatch			
• Interconnect/fatigue			
• Packing factor			
• UV degradation			
Trapped particle degradation	0.850	0.1150	311.7
° Solar cells			
Array power distribution	0.96	0.1104	299.2
Rotary joint power distribution	0.98	0.1082	293.26
Incident solar power ≈ 1353 W/m^2 × 2 = 2706 W/m^2 Solar panel area = 900,000 m^2 Power from array = 900,000 2706 0.1354 = 329.75 MW			

The blankets were found to be capable of delivering 366.4×10^6 W/m^2 and, consequently,

$$P_{SB} = 900{,}000 \times 366.4 = 329.75 \text{ MW}. \tag{1}$$

The power available at the thruster modules, P_{TH}, is given by

$$P_{TH} = 9.94 \times 329.75 = 310.22 \text{ MW}. \tag{2}$$

However, because of degradation from geomagnetically trapped particles, and periodic annealing, the average power is assumed to remain near 85.0 percent of the above values. Thus, with degradation

$$P_{SB} = 280.3 \text{ MW}, \tag{3}$$

and

$$P_{TH} = 263.7 \text{ MW}. \tag{4}$$

Nevertheless, the size of the thruster array is based on the maximum module power, 310.22 MW, which is available at the beginning of mission. This appears to be a weight penalty, but as the mission proceeds the idle thrusters become usable spares.

Thrust and Mass

The gross initial or beginning-of-life (BOL) departure mass in LEO is given by

$$M_P^U = M_P + M_F,$$

where

$$M_P^U = \text{Total ascent propellant required between LEO and GEO}, \tag{5}$$

and

$$M_F = \text{Final mass in GEO}.$$

Each thruster module requires an input power of about 1.20 MW. The number of thrusters is then found from

$$N_{TH} = \frac{310.22}{1.2} = 259.$$

The maximum total thrust, with an assumed 13-N thrust per module, is 3364 N which occurs at BOL. The average thrust, based on 220 thrusters operating at rated power, is 2859 N. The thrust/weight ratio for the assumed 15 percent mean degradation is 5.4219×10^{-5} N/kg.

Thus,

$$gM_i = 2859/(5.4219 \times 10^{-5}) = 5.273 \times 10^7 \text{ kg (force)} \tag{6}$$

or

$$M_i = 5.377 \times 10^6 \text{ kg (mass)}. \tag{7}$$

The propellant requirements are

$$M_P^U = 0.03475\ M_i = 186{,}864\ \text{kg} \qquad (8)$$

$$M_P^D = 60{,}636\ \text{kg} \qquad (9)$$

and

$$M_{PT} = 247{,}500\ \text{kg} \qquad (10)$$

The masses of propellant tanks and lines are assumed to be 20 percent of the total propellant mass, i.e., 49,500 kg.

A summary of COTV component masses, along with the estimated payload mass, is shown in Table 5.3-5.

Table 5.3-5. OTV Component and Payload Mass Breakdowns (15% Degradation)

Component	Mass (kg)
Solar array and reflector	304,057
Power distribution	372,109
Thruster modules	31,080
Propellant (up)	186,864
Propellant (down)	60,636*
Propellant tanks and lines	49,500
Structure	62.046
Rotary joint	321,885
A/C	53,770
OTV mass (BOL)	1,441,947
Payload mass	3,935,053
BOL mass in LEO	5,377,000
*Includes a margin of 8733 kg (14.4%)	

Table 5.3-6 presents COTV data for mean solar blanket degradations of o, 15, and 20 percent. The COTV fleet size required to transport one SPS, having a mass of 37×10^6 kg, to GEO, ranges from 7 for zero degradation, 10 for 15% degradation, and 11 for 20% degradation.

The dedicated electric OTV configuration, shown in Figure 5.3-4, was sized to accommodate a payload capability of approximately 4×10^6 kg. The structural configuration is essentially the same as employed for the SPS, and is sized to produce approximately 264 megawatts at the thruster modules.

The thruster array is suspended by cables and located at the vehicle c.g. The thruster array is comprised of six subarrays (6×30 m), each of which is capable of being packaged in the winged-HLLV cargo bay. Approximately 259 one-meter electric thrusters are required for primary thrust. Additional

Table 5.3-6. COTV Parameters for 0%, 15%, and 20% Mean Solar Blanket Degradation

Solar Blanket Power (MW)	Power at Thruster Module (MW)	Assumed Mean Degradation (%)	Required Mean Thrust (%)	OTV Mass (kg)	P/L Mass (kg)	Propellant and Tanks (kg)	GEO/ Winged HLLV Trip* (kg)
330.0	310.3	0	3363	1,700,658	5,737,242	393,690	63,818
280.5	263.9	15	2859	1,441,947	3,935,053	284,625	60,539
264.0	284.4	20	2691	1,316,005	3,443,995	277,157	59,895

*100% packaging

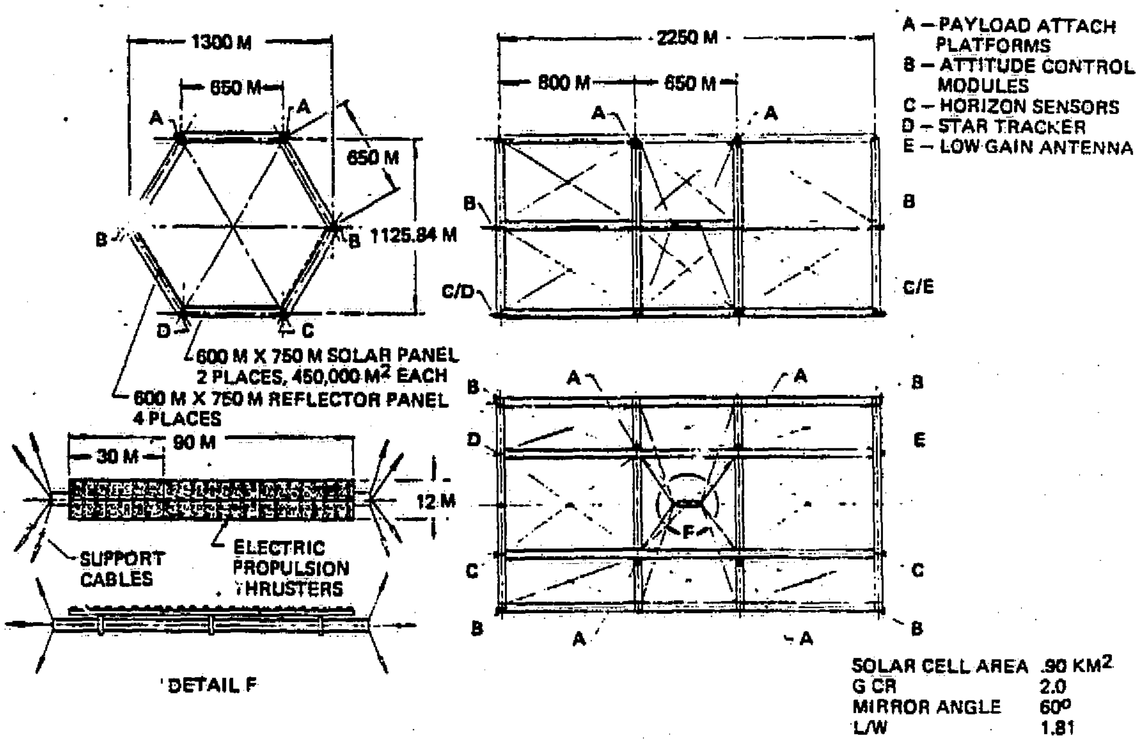

Figure 5.3-4. Electric OTV Concept

attitude control thruster packages are located at the structural extremities. Primary thrust vector control is accomplished by a slip ring joint identical to the type used for SPS antenna orientation.

6.0 PERSONNEL TRANSPORT SYSTEMS

6.0 PERSONNEL TRANSPORT SYSTEMS

Personnel transport systems are required to support the space construction operations described in Section 9 of this volume. The three primary elements of the personnel transport system identified are an earth launch vehicle, an orbital transfer vehicle, and a crew and resupply module.

6.1 EARTH LAUNCH VEHICLE

If an unmanned ballistic HLLV of the type described in Section 4.1 is employed for SPS cargo transport to LEO, it will be necessary to utilize the Space Shuttle Transportation System (STS) or a derivative thereof for personnel transport from earth-to-LEO. On the other hand, if the winged configuration described in Section 4.2 is employed in the SPS program, that vehicle would also satisfy the requirement for personnel transport to LEO. The HLLV selected for this phase of the study is the latter or winged HLLV configuration.

6.2 ORBITAL TRANSFER VEHICLE

The orbital transfer vehicle selected for personnel transport to and from GEO is the common stage chemical OTV described in Section 5.1.2. This configuration would require three winged vehicle flights (i.e., one HLLV flight for each stage and one HLLV flight for the crew and resupply module discussed below) for each personnel orbital transfer mission. The three elements would be assembled in LEO and later recovered in LEO after the mission and returned to earth on subsequent winged HLLV flights.

6.3 CREW AND RESUPPLY MODULE

In Section 9 a construction sequence is developed that requires a crew rotation every 90 days for crew complements in multiples of 48. A crew and resupply module (CRM) was synthesized on this basis. Based on previous studies of passenger modules for an orbiting lunar station, lunar surface base, geosynchronous station, and LEO space station, a parametric sizing curve for passenger modules was developed (Figure 6.3-1). These data indicated that for a crew complement of 48 persons, the module would weigh approximately 200 kg (440 lb) per man, or 9,600 kg. Comparable data were extracted from these studies for consumables, passenger/personal effects, in-transient consumables, crew module, resupply module, and on-orbit habitable module spares. The resultant logistics profile for a 48-man contingent at geosynchronous orbit for 90 days is presented in Table 6.3-1.

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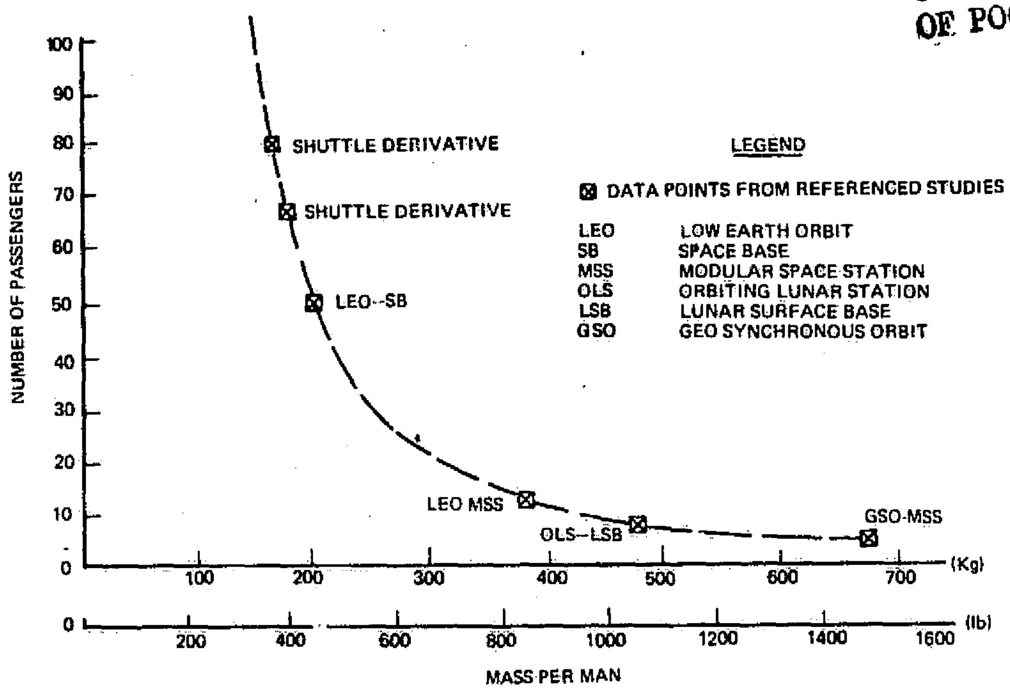

Figure 6.3-1. Passenger Module Mass Trend
(No. of Passengers Versus Mass/Man)

Table 6.3-1. Crew Rotation/Resupply Logistics Profile

Item	Factor	Up Payload (kg)	Down Payload (kg)
Personnel/personal effects	48 men X 110 kg/man	5,280	5,280
On-orbit consumables	3.6 kg/man-day X (48 men) X (90 days)	15,550	-
Consumables containers	10% of consumables	1,555	1,555
Passenger module	200 kg/man X 48 men	9,600	9,600
Resupply module	20% of consumables	3,110	3,110
OTV crew module*	Self-sufficient - 2-man crew	2,000	2,000
Total		37,095 (81,600 lb)	21,545 (47,400 lb)

*Considered as integral part of passenger modules

A conceptual layout of the CRM is shown in Figure 6.3-2. It was assumed that a command module area would be required to monitor and control OTV performance during crew rotation flights. This function was incorporated in the forward section of the passenger module as shown. Spacing and layout of the passenger module is comparable to current commercial airline practice. A nominal packing density of 160 kg/m^3 (10 lb/ft^3) was assumed for resupply consumables. It was assumed the resupply modules would be exchanged each mission. While at GEO, the resupply module could be used as the consumables storage module. Thus, multiple access aisles also were included in the sizing of the resupply module. A gross packing density of 93 kg/m^3 (6 lb/ft^3) resulted, which allows for a large growth factor.

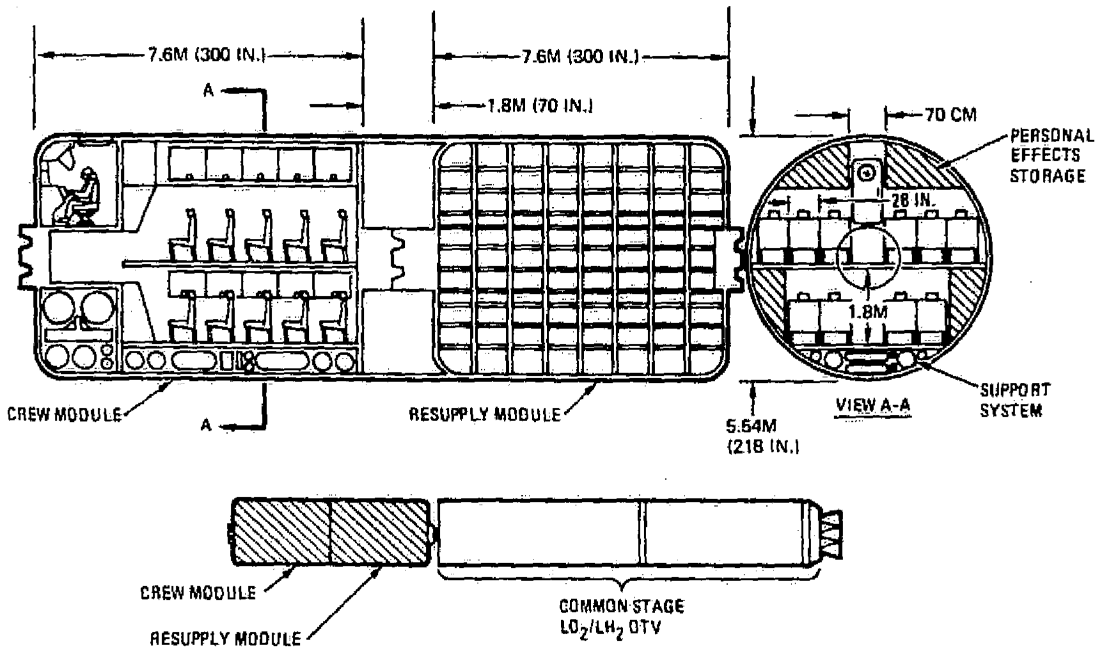

Figure 6.3-2. Crew and Resupply Module

7.0 GROUND OPERATIONS

7.0 GROUND OPERATIONS

A primary objective is to identify the necessary resources to meet the launch rate requirements of the selected HLLV configurations based upon payload mass flow requirements. The selected HLLV configurations are the two ballistic configurations described in Section 4.1, and the winged configuration described in Section 4.2. The winged vehicle and small ballistic HLLV require 16 launches/day, and the large ballistic HLLV requires 4 launches/day.

Timeline analyses were performed to evaluate turnaround time requirements for flight hardware, launch sites, processing/integration facilities, support equipment, and recovery vessels. From these timelines, requirements were formulated for numbers of launch sites, launch vehicles, support equipment, and facilities. The small HLLV requires 33 launch sites, 61 first-stage elements, and 70 second-stage elements in the turnaround cycle. The large HLLV requires 10 launch sites, 22 first-stage elements, and 23 second-stage elements in the turnaround cycle. Recovery operations were evaluated to assess feasibility and degree of complexity. A special recovery type vessel is deemed necessary to effectively accomplish recovery of the HLLV stages. The ship requires dry-dock-type capability, heavy crane capacity, high maneuverability, size sufficient to hold three launch vehicle stages, and the ability to recover the launch vehicles in moderate seas (wave height <10 ft).

Refurbishment operations were identified and modified from Space Shuttle and Shuttle growth requirements. Launch pad refurbishment, assuming minimal damage and routine maintenance, is estimated to be 30 hours. The MLP is assumed to be refurbished on the launch pad in parallel with pad refurbishment. Vehicle refurbishment is based on negligible water impact damage and minimal salt water contamination of the engines. Engine repair or replacement requires that a vehicle go "off line" from the normal turnaround cycle.

Acoustic hazard analyses were performed to determine typical launch site, processing facility, and port facility geographical separation requirements. Because of the sound pressure levels estimated for launch vehicles of this size, the land distance required is 45 km for the large HLLV and 55 km for the small HLLV, preferably along a coastal area. General facility plans are presented that would allow parallel stage processing/vehicle integration to support the prescribed launch rates.

Weather conditions at the launch site and sea conditions at the recovery area were investigated for overall impact on the traffic model. Weather for a KSC launch will have an effect on the launch rate which might be compensated for by higher launch rates, or by shifting from a five- to a seven-day work week to make up for missed launch opportunities due to inclement weather.

The basic ground rules and assumptions used for the ballistic HLLV ground operations analyses are presented in Table 7.0-1.

Table 7.0-1. Ground Rules and Assumptions

- ✓ LAUNCH RATES - SMALL HLLV (91×10^3 KG PAYLOAD), 16/DAY
  - LARGE HLLV (454×10^3 KG PAYLOAD), 4/DAY
- ✓ 15 ORBITS/DAY (REPEAT TRACE)
- ✓ LAUNCH OPPORTUNITY (EACH LEO STATION/DAY)
  FIRST PASS
  THIRD PASS (3:20 HR AFTER FIRST L/O TIME)
- ✓ LEO STATIONS - 4 @ 90 DEG
- ✓ IMPACT/RECOVERY AREA
  NOMINAL DOWNRANGE DISTANCE (NMI) - 170 (1ST STAGE), 90 (2ND STAGE)
  ELLIPTICAL FOOTPRINT (NMI) - 5×9 (1ST STAGE), 8×20 (2ND STAGE)
- ✓ 300-MISSION LIFE/VEHICLE
- ✓ 20 MISSIONS BETWEEN ENGINE CHANGEOUT/OVERHAUL
- ✓ SHUTTLE PROPELLANT FILL RATES - 12,000 GPM, LH_2 ; 1250 GPM, LO_2
- ✓ LOAD PROPELLANTS SEQUENTIALLY
- ✓ SHUTTLE PROPELLANT VENT RATES
- ✓ STAGE TWO IMPACTS APPROX. 24 HOURS AFTER LAUNCH
- ✓ 130 DB ACOUSTIC NOISE LEVEL BETWEEN LAUNCH SITES
- ✓ 120 DB ACOUSTIC NOISE LEVEL FROM THE LAUNCH SITES TO THE PROCESSING/INTEGRATION FACILITY AND PORT (ACCEPTABLE LIMIT FOR DAILY OPERATION
- ✓ ONE SPARE VEHICLE IS LAUNCH-READY TO BACK UP EVERY LAUNCH ATTEMPT
- ✓ TWO RECOVERY SHIPS/STAGE ON STATION FOR SMALL HLLV PICKUP; ONE SHIP/STAGE FOR LARGE HLLV PICKUP
- ✓ THREE STAGES RECOVERED PER SHIP CYCLE
- ✓ RECOVERY SHIP AVERAGE SPEED APPROX. 15 KNOTS
- ✓ FIVE-DAY WORK WEEK, 3 SHIFTS/DAY
- ✓ VERTICAL POSITION STAGE RECOVERY AND STOWAGE ON RECOVERY SHIP
- ✓ PARALLEL UNLOADING OF STAGES
- ✓ RCS USES SAME PROPELLANTS AS STAGE
- ✓ STAGE TANKS ARE PROTECTED FROM SALT WATER CORROSION
- ✓ MINIMAL SALT WATER CONTAMINATION OF ENGINES

The winged vehicle is capable of operating from any launch/landing site that can presently accommodate 747-type aircraft, provided that LO_2/LH_2 service is available. It would jettison its takeoff gear at approximately 1000 ft, with recovery by parachute. Flight to and from orbit would require a total of approximately six hours, with two hours on orbit to unload cargo. The winged HLLV would return to its landing site--preferably an isolated locale where spacing of facilities would present no problem. At this site, the winged HLLV would off-load cryogenics, be serviced and maintained, have cargo and propellants loaded, and be ready for reflight. Turnaround time for the vehicle is estimated to be 43.25 hours, based on modified aircraft and Shuttle processing times. Thirty winged HLLV's could satisfy the satellite requirements of 80 flights/week.

7.1 BALLISTIC HLLV OPERATIONS

The launch and operational flow requirements are based upon significant improvements in current launch vehicle handling and processing technology. Launch and operations are minimized due to the high launch rate requirements and to minimize potential interferences between launch pads.

7.1.1 LAUNCH SCHEDULE

Each of the four low earth orbit stations has two launch opportunities per day. The small ballistic HLLV requires three launch vehicles flight-ready for each launch opportunity (one vehicle serves as backup). The backup vehicle then is a prime vehicle at the next selected launch opportunity. The ballistic HLLV launch schedule is shown in Figure 7.1-1. The launch opportunities for any one particular station are 3 hours and 20 minutes apart, and the initial launch opportunity for the different stations is space six hours apart.

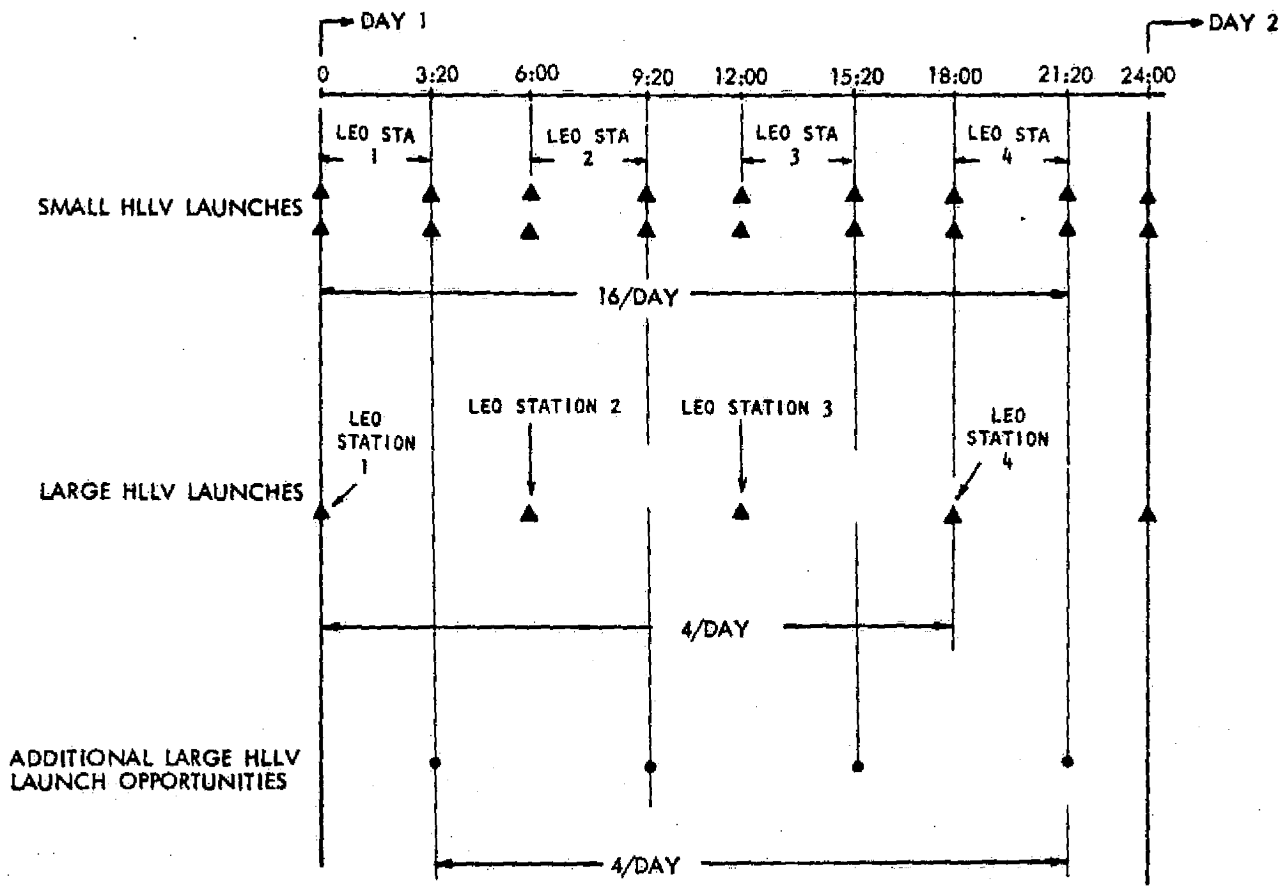

Figure 7.1-1. Ballistic HLLV Launch Schedule

The large ballistic HLLV also has two launch opportunities to each low earth orbit station per day, but requires only one launch/day/station (one vehicle also stands by launch-ready for each scheduled launch).

7.1.2 OPERATIONAL FLOW

The HLLV operational flow cycle is shown in Figure 7.1-2. This flow commences at the vehicle integration facility where Stage 1, Stage 2, and the payload are mated to the mobile launch platform (MLP). The integrated test is performed and the vehicle then is transported to one of the launch sites. The pad/MLP interfaces are mated and prelaunch verification tests performed. Upon completion of testing, the vehicle propellants are sequentially loaded. The vehicle is then closed out and readied for countdown and launch. At this point, there are three simultaneous operations in the flow process--launch pad refurbishment, MLP refurbishment, and recovery ship operations.

The launch site and MLP require LN_2 inerting, a damage/safety inspection, propellant flushing/purging, and general refurbishment. The MLP is then transported back to the integration facility for new vehicle buildup, and the launch pad is ready to accept a new MLP and vehicle. While MLP/pad refurbishment is in progress, the recovery ships are retrieving spent stages. The ships remain on station until three spent stages are retrieved and then return to port. The docking facility design permits simultaneous removal of at least two stages simultaneously from a single recovery ship. The stages are then loaded on vehicle-type transporters and moved to their processing facility. The stage processing facility is located within the same building as the integration facility in order to minimize handling. The stages are positioned in the processing stands for engine flushing/purging, battery replacement, subsystem verification, and inspection. At the completion of refurbishment/verification, the stages are ready to move into the integration cells to start another vehicle cycle. Payloads are processed in a separate facility, and enter the integration facility as required.

7.1.3 TIMELINE ANALYSIS

Detailed timeline flows were prepared for the major operations in a turnaround cycle. Both vehicles are two-stage, water recoverable. The recovery vessels, representing new technology, have floating dry-dock capability with sea doors and flooding compartments front and rear. Stages are recovered in the vertical position (three per ship). Simultaneous vehicle unloading is performed and the stages are moved by ground transporters to their processing facility. The stage processing stations and vehicle integration stations are located in a common building to minimize transport. The total vehicle (on a mobile launcher platform) is transported to the launch site by a crawler transporter.

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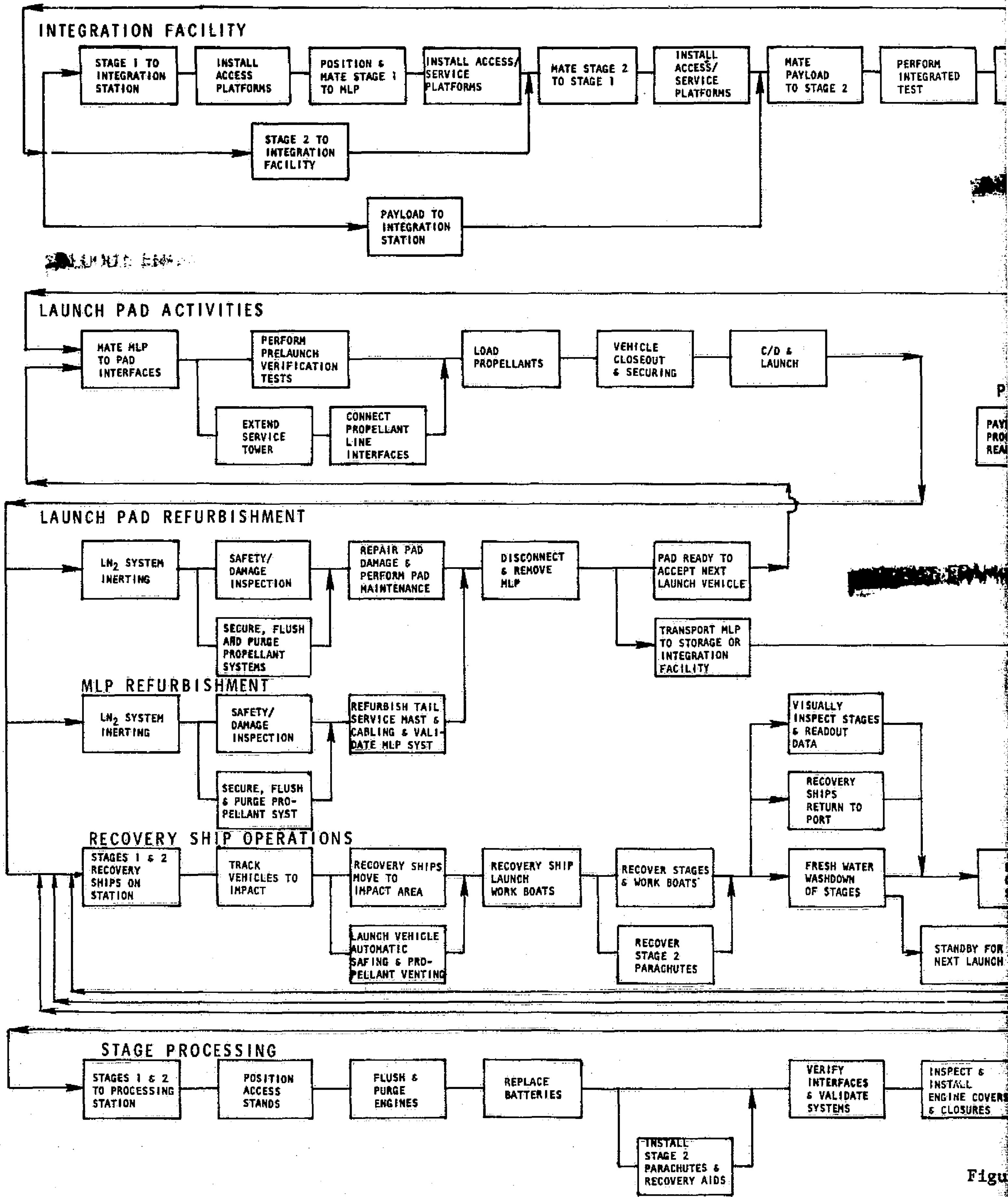

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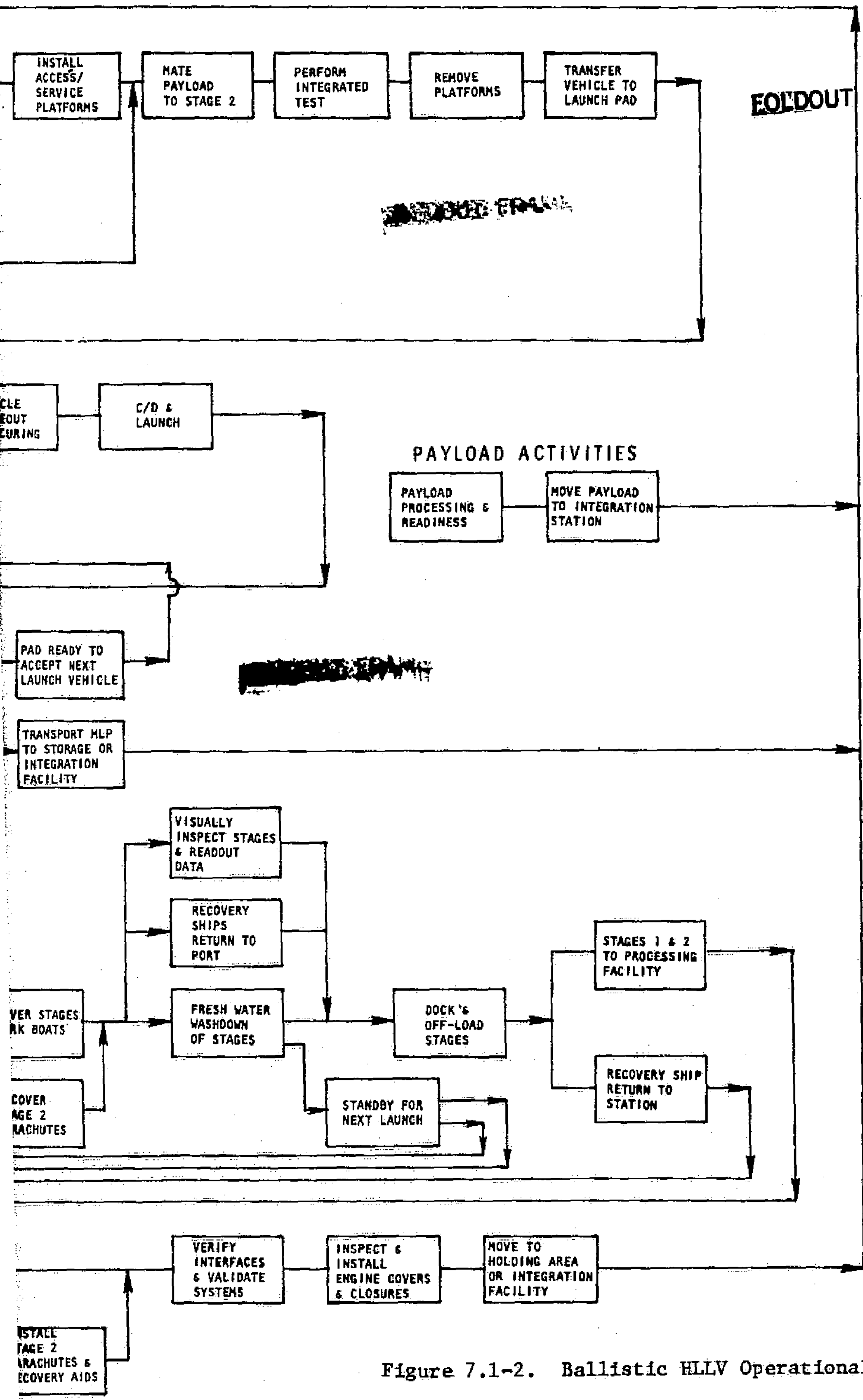

Figure 7.1-2. Ballistic HLLV Operational Flow

7.2 BALLISTIC HLLV - SMALL (91,000-kg PAYLOAD)

7.2.1 INTEGRATION AND CHECKOUT OPERATIONS

The stages and payload arrive at the integration facility from their respective processing stations, Figure 7.2-1. The MLP has already been positioned and is ready for vehicle buildup. Each stage has undergone subsystem refurbishment and checkout in its processing station. The vehicle and payload are assembled and an integration test performed. Following data review and acceptance, the vehicle is closed out and prepare for movement to the launch site. The integration station is occupied for a total of 21 hours.

7.2.2 LAUNCH AND OPERATIONS

Transport to the launch pad on the crawler transporter requires eight hours. Once at the launch site the mobile launcher platform is positioned and secured to the pad, Figure 7.2-2. All pad/MLP interfaces are connected and prelaunch verifications performed. Propellants (RP-1/LO_2LH_2) are sequentially loaded through the tail service mast and service tower lines. Topping of the LO_2/LH_2 continues until just prior to liftoff. The vehicle lifts off 17 hours after arriving at the launch site.

7.2.3 LAUNCH PAD/MLP REFURBISHMENT

The launch pad and MLP are concurrently refurbished after launch, Figure 7.2-3. The LH_2 systems are first inerted, then the safety/damage inspection teams are allowed to enter. During the inspection time the propellant systems are secured, flushed, and purged as required. The pad/MLP propellant interfaces are then disconnected. Normal pad/MLP refurbishment of structure, cabling, and other systems is then performed. Thirty hours after launch, the MLP is removed and the pad is ready to accept another vehicle. MLP transport to the integration facility requires seven hours. The MLP is installed in the integration facility and interfaces verified in preparation for vehicle buildup. In the same time period that the launch pad and MLP are being refurbished, spent Stages 1 and 2 are being retrieved at sea.

7.2.4 RECOVERY OPERATIONS

The recovery ships (dry-dock capability, front and rear) are on station near the stage impact area and track the stages to impact. Since the small HLLV has simultaneous launches, two recovery ships are required for each stage impact area; this minimizes salt water exposure time for the stages. Stage 1 drops off on a ballistic path, is oriented by RCS engines, and makes a soft ocean impact with the aid of three SSME engines. After splashdown, the stage will automatically safe all pyrotechnics and start venting remaining cryogenic propellants. Hydrogen will be vented first, then oxygen, with a short delay interval between propellants. Normally, propellant venting of the first and second stage will be complete by the time the recovery ship arrives in the impact area. Work boats will be launched which will assist in recovery operations. Stage 1 recovery operations are shown in Figure 7.2-4.

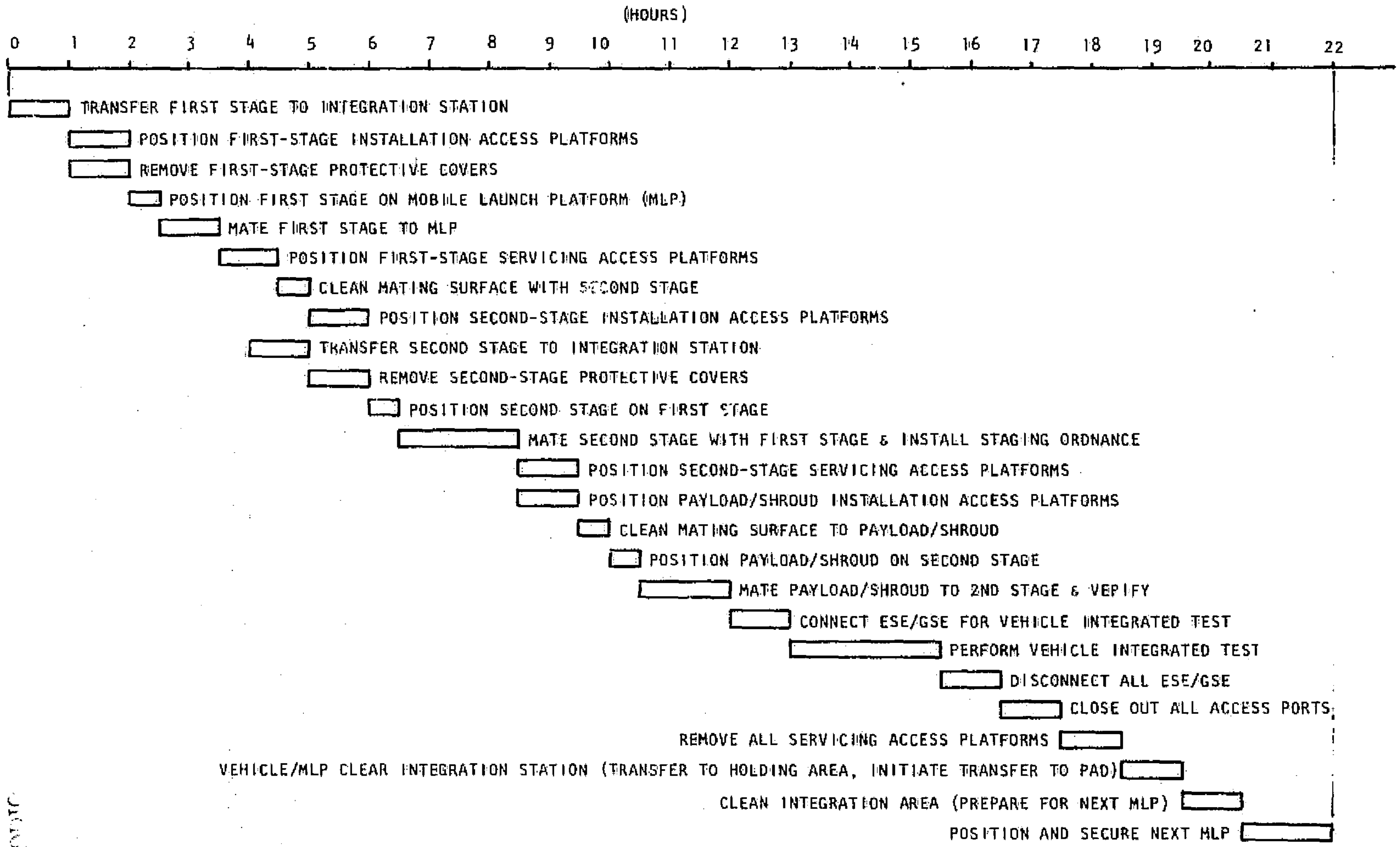

Figure 7.2-1. Small Ballistic HLLV Integration and Checkout Operations

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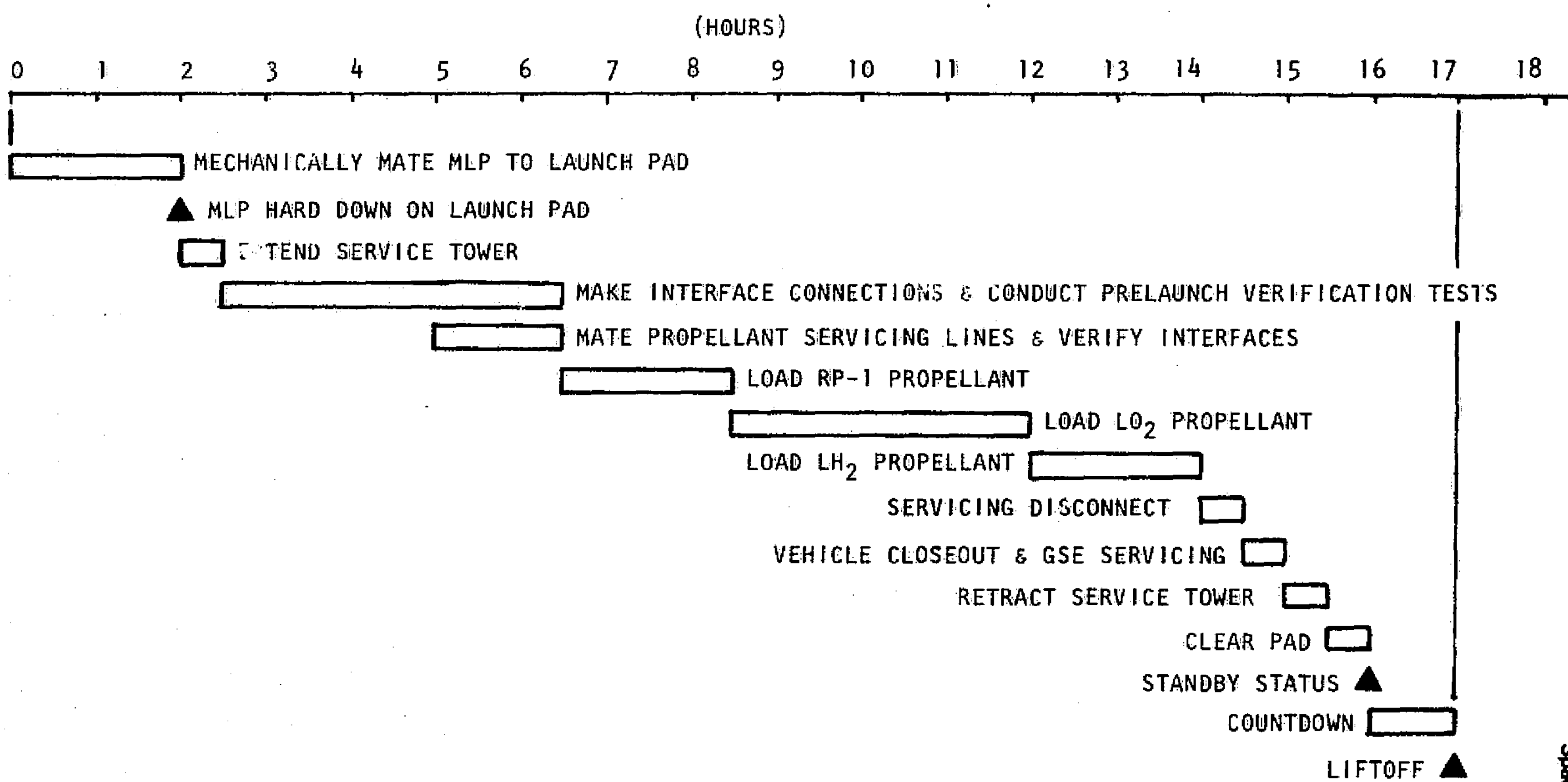


Figure 7.2-2. Small HLLV Launch Pad Operations

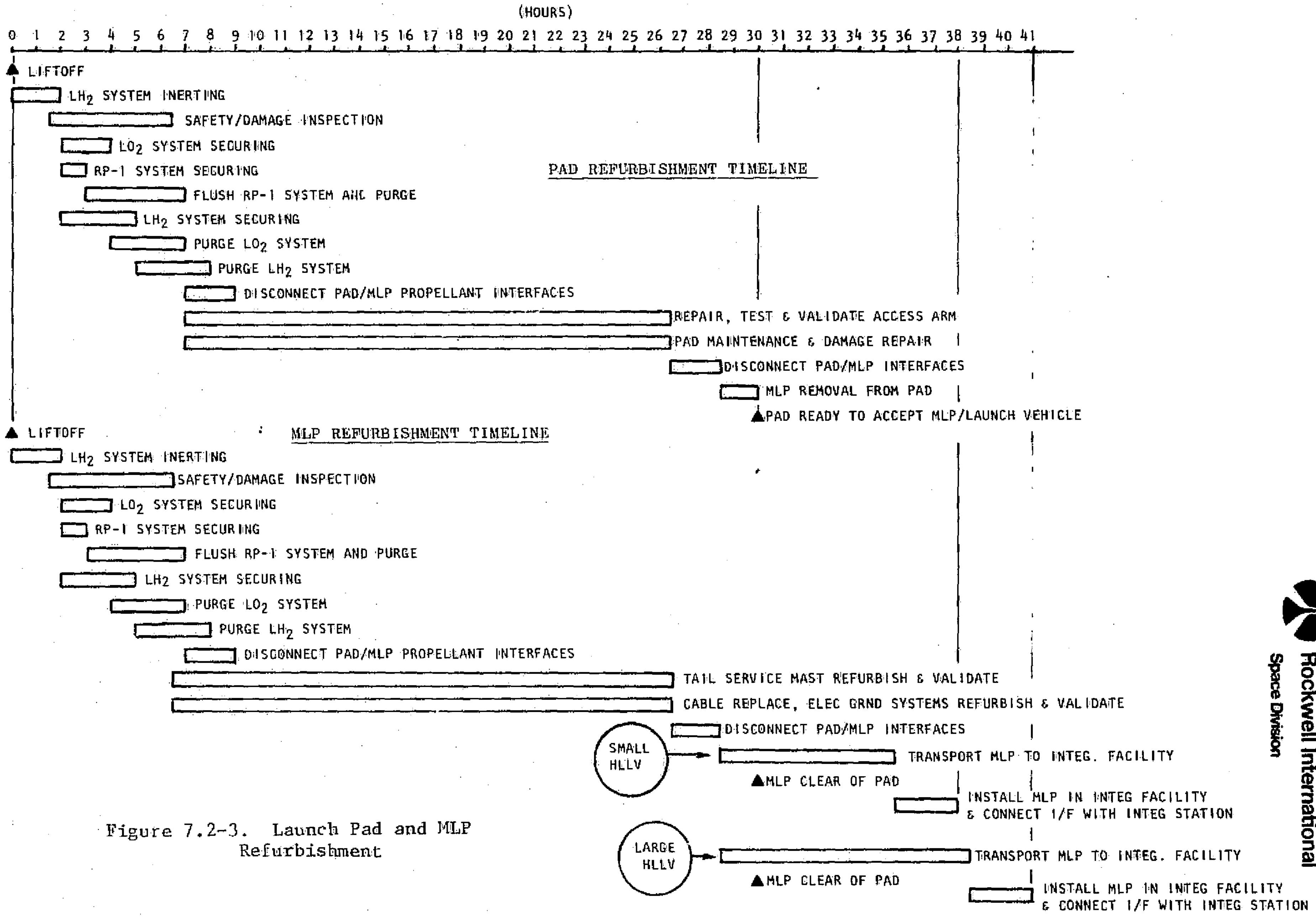


Figure 7.2-3. Launch Pad and MLP Refurbishment

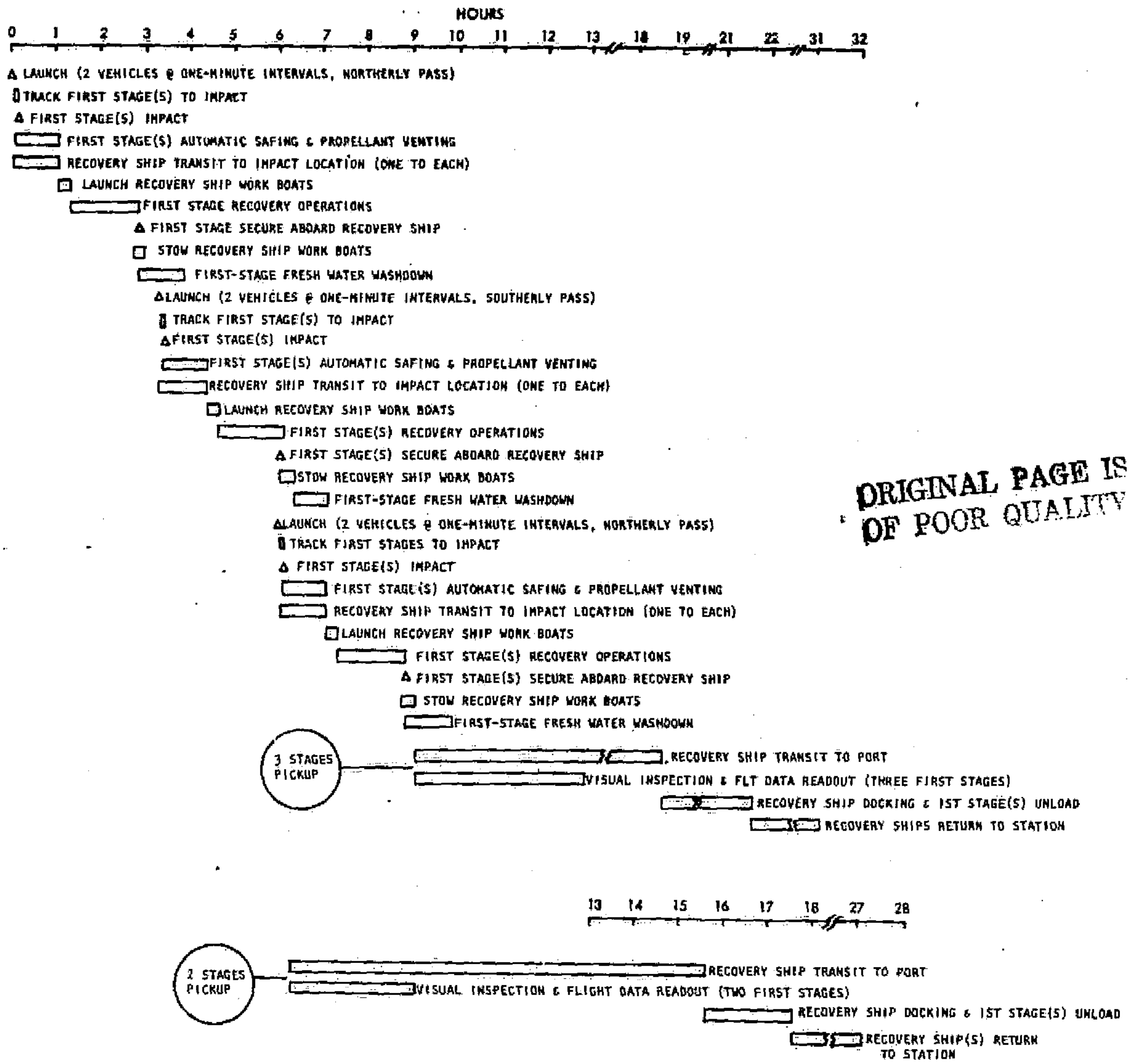


Figure 7.2-4. First-Stage Recovery Operations - Small HLLV

The recovery vessels remain on station until three stages have been retrieved. Stage 1 impacts approximately 170 nmi down range, requiring 11.5 hours for the recovery ship to reach port. Total cycle time on a Stage 1 recovery ship is 35 hours.

Stage 2 recovery takes longer than Stage 1 due to retrieval of the added parachute systems employed for landing. The Stage 2 recovery timeline, Figure 7.2-5, reflects the longer recovery time, but due to the closer impact point (90 nmi), a Stage 2 recovery ship has a recycle time of only 25.5 hours. During the time the recovery ships are steaming to port, the vehicle stages are given a fresh-water washdown. Damage inspection is performed, and flight performance data read-out accomplished. In this way, determination can be made prior to reaching port whether a stage returns to the in-line processing

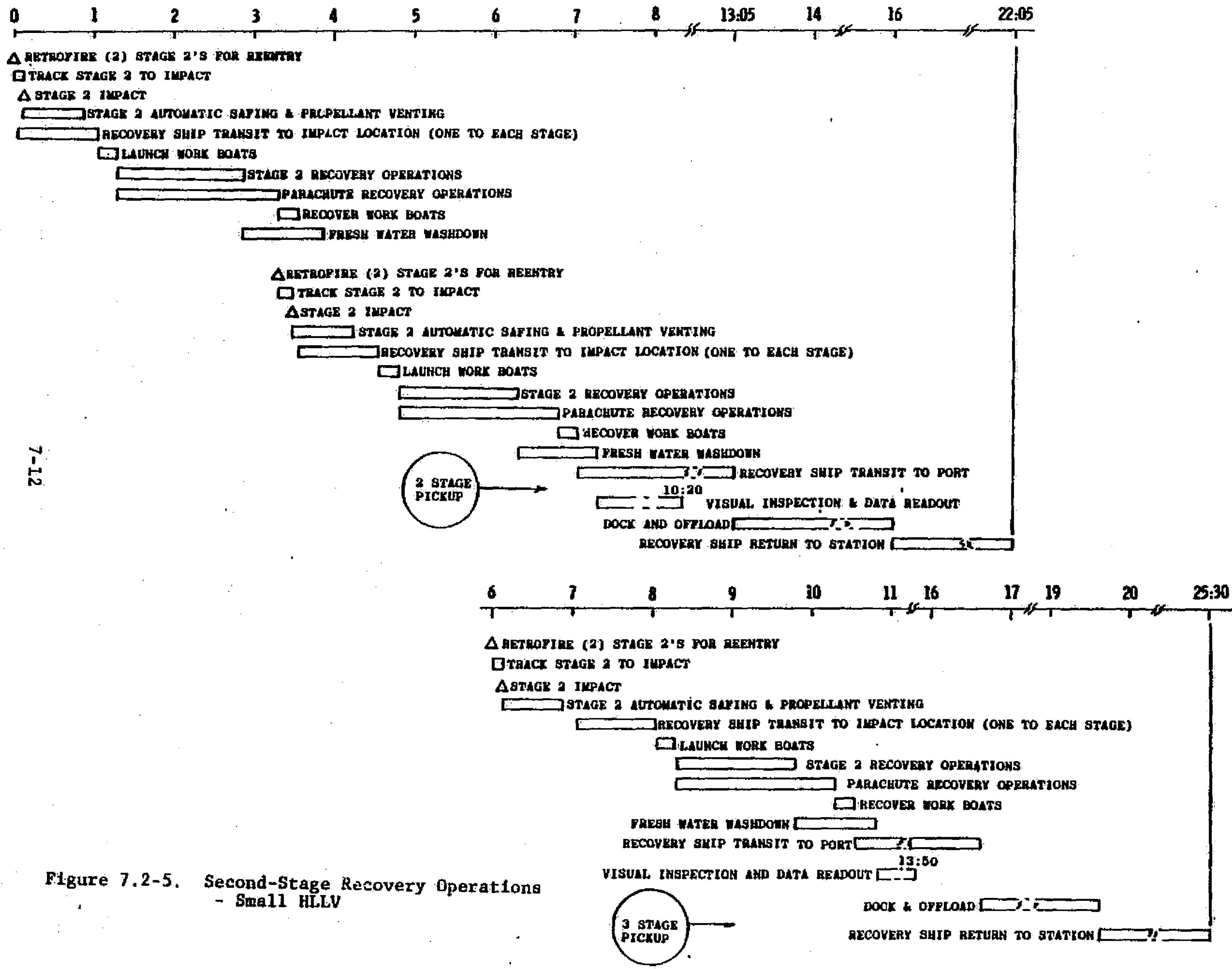


Figure 7.2-5. Second-Stage Recovery Operations - Small HLLV

facility, or to some other facility, for engine changeout or structural repair. The stages are moved from the port facilities to their processing stations on individual transporters.

7.2.5 STAGE PROCESSING

At the processing facility, the individual stages are positioned in checkout cells for refurbishment and reverification, Figure 7.2-6. Engines are flushed, purged and dried, batteries replaced, parachute systems replaced (Stage 2 only), and subsystems verified. Stage 1 requires 16 hours, while Stage 2 requires 13.5 hours, in the processing facility. The stages are then moved to the adjacent integration facility for vehicle assembly.

7.2.6 PAYLOAD ACTIVITIES

Payload preparation will be accomplished in an off-line facility from launch vehicle processing. The payload and shroud will be assembled and arrive at the integration facility as an integral unit for placement on the launch vehicle.

7.2.7 TURNAROUND OPERATIONS

The small HLLV turnaround operations summary, Figure 7.2-7, reflects a combination of all the individual facility, vehicle, and support equipment timelines. Stage 2, due to its 24 hours on orbit prior to reentry, has the longest turnaround time of all the elements (100.5 hours). Stage 1 has a turnaround time of 87 hours. This is partly due to its slightly longer processing time and recovery at a greater distance out to sea. The launch pad is recycled in 47 hours, based on minimal launch damage and the need for only routine refurbishment. For the timelines shown, the small HLLV is capable of supporting 16 launches per day on a 24-hour/day, 5-day/week basis.

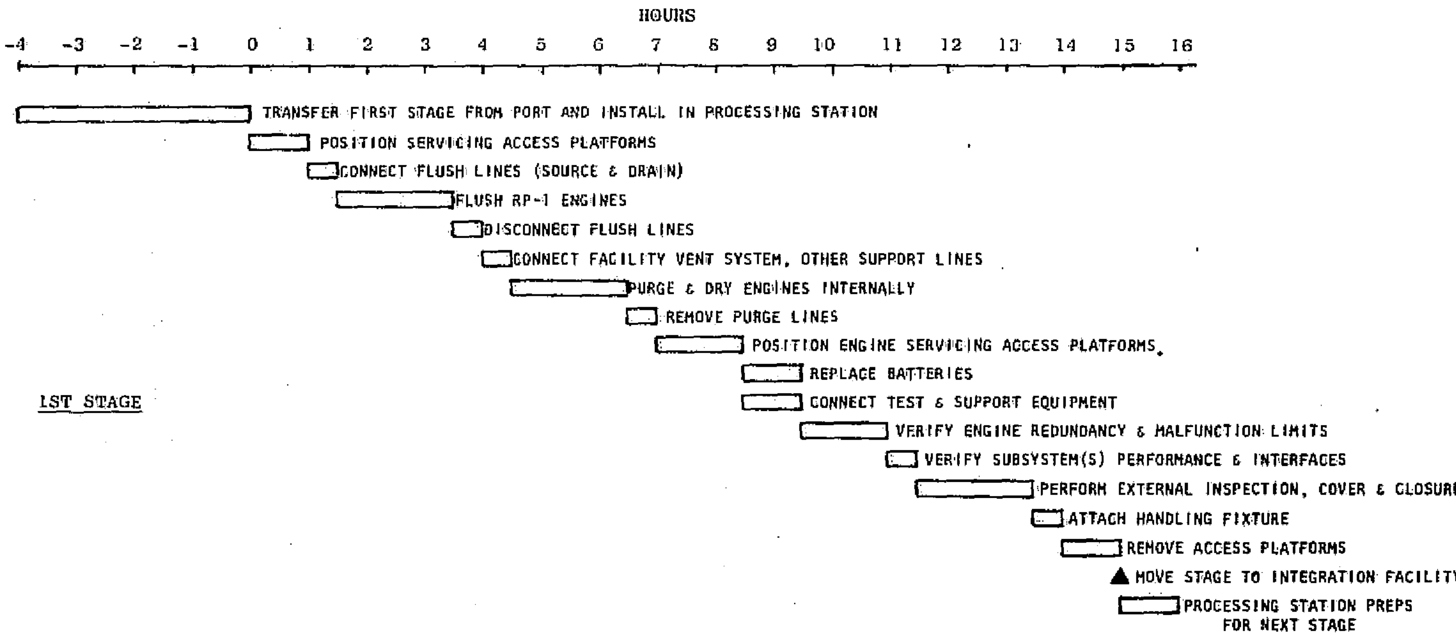
7.3 BALLISTIC HLLV - LARGE (400,000-kg PAYLOAD)

7.3.1 INTEGRATION AND CHECKOUT OPERATIONS

The large HLLV integration operations, Figure 7.3-1, are similar to those of the small HLLV, but require a longer time to accomplish because of the larger number of engines. The launch vehicle spends a total of 18.5 hours in this facility. The crawler transporter carries the MLP and launch vehicle to the launch site in 11 hours.

7.3.2 LAUNCH PAD OPERATIONS

At the launch site, the MLP is positioned and mated to the pad, Figure 7.3-2. Interfaces are connected and all prelaunch verification tested conducted. The vehicle requires a total of 21 hours at the launch site, in comparison to only 17 hours for the small HLLV. The additional time is accounted for in interface connections, testing, and loading the larger quantity of propellants.



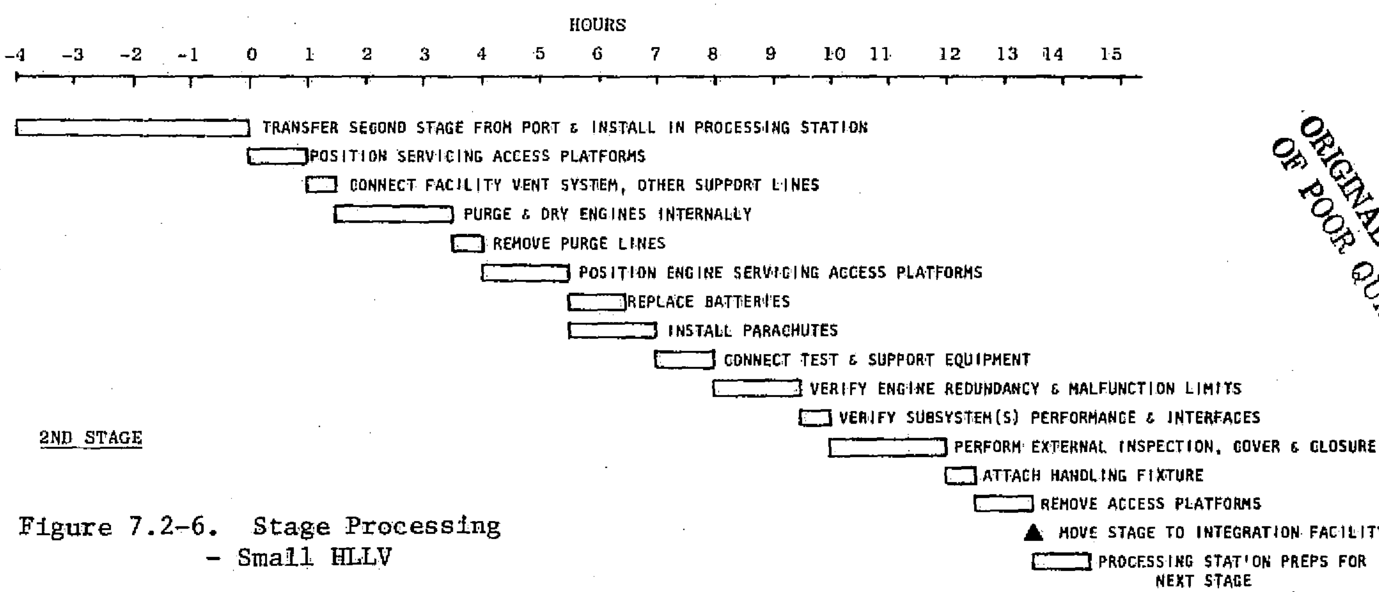


Figure 7.2-6. Stage Processing
- Small HLLV

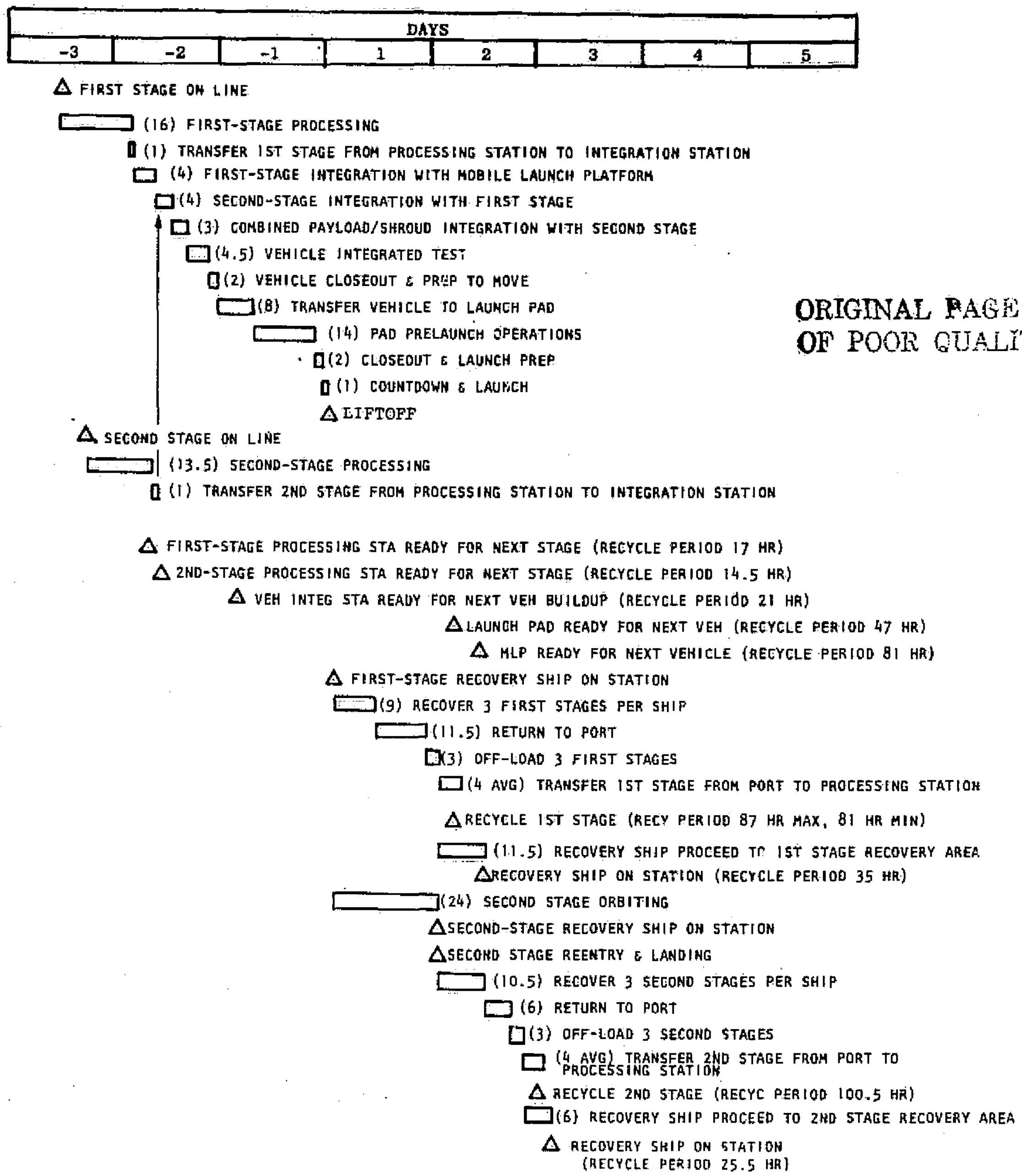


Figure 7.2-7. Turnaround Operations Summary
- Small HLLV

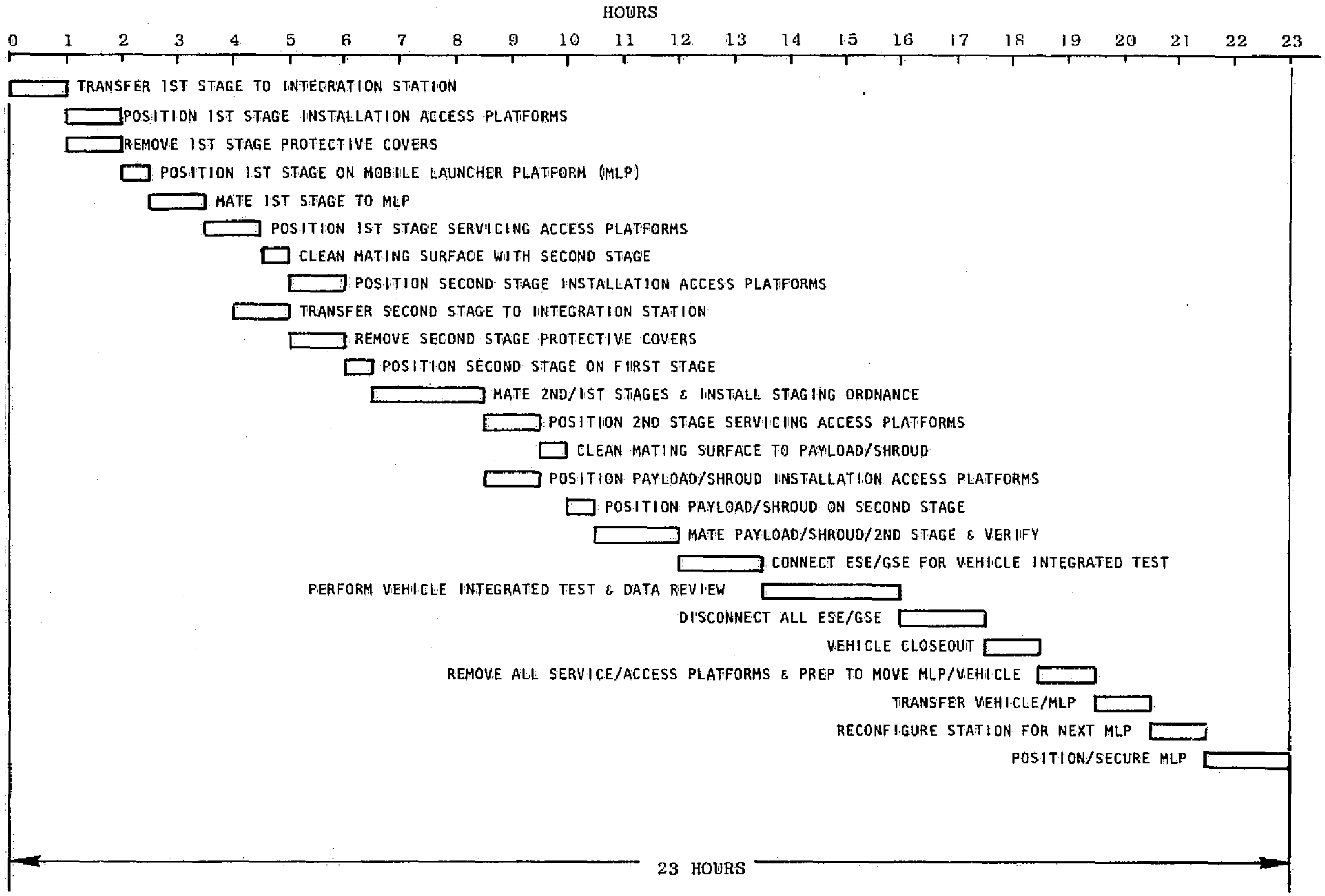


Figure 7.3-1. Integration and Checkout Operations - Large HLLV

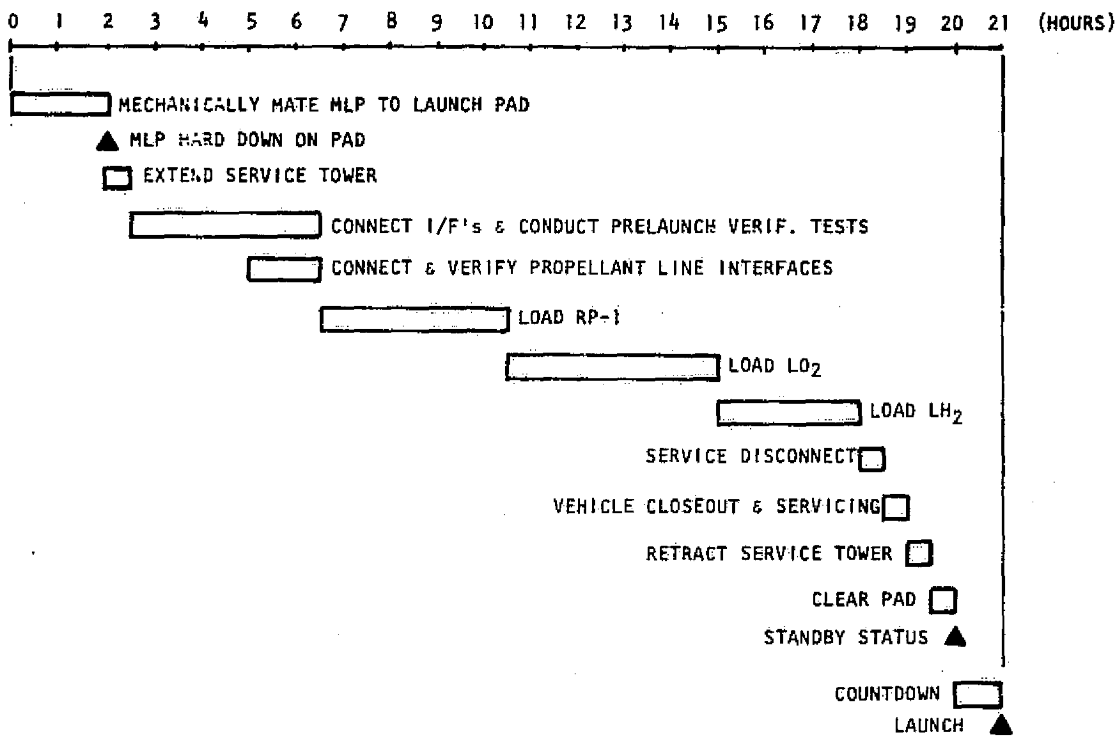


Figure 7.3-2. Launch Pad Operations - Large HLLV

7.3.3 LAUNCH PAD/MLP REFURBISHMENT

The large HLLV launch pad and MLP are refurbished along the same timelines as the small HLLV (Figure 7.2-3). Pad turnaround is accomplished in 51 hours from receipt of a launch vehicle until it is again ready to accept another vehicle. Transport of the MLP back to the integration facility and installation/verification in the integration cell requires a total of 12.5 hours. Concurrent with pad/MLP refurbishment, the recover ships are retrieving spent stages.

7.3.4 RECOVERY OPERATIONS

The large HLLV recovery ships are of the same floating dry-dock type, but necessarily larger to handle the larger stages. Since this vehicle has only one scheduled launch every six hours, only one first-stage and one second-stage recovery ship is required on station at any one time. The recovery ship proceeds to the impact area, but has to stand by for approximately one hour until completion of residual propellant venting. Stage 1 recovery operation timelines are detailed in Figure 7.3-3. After retrieving three stages, the recovery vessel returns to port while washing the stages down and reading out flight data. Recycle time on a Stage 1 recovery ship is 42-1/4 hours.

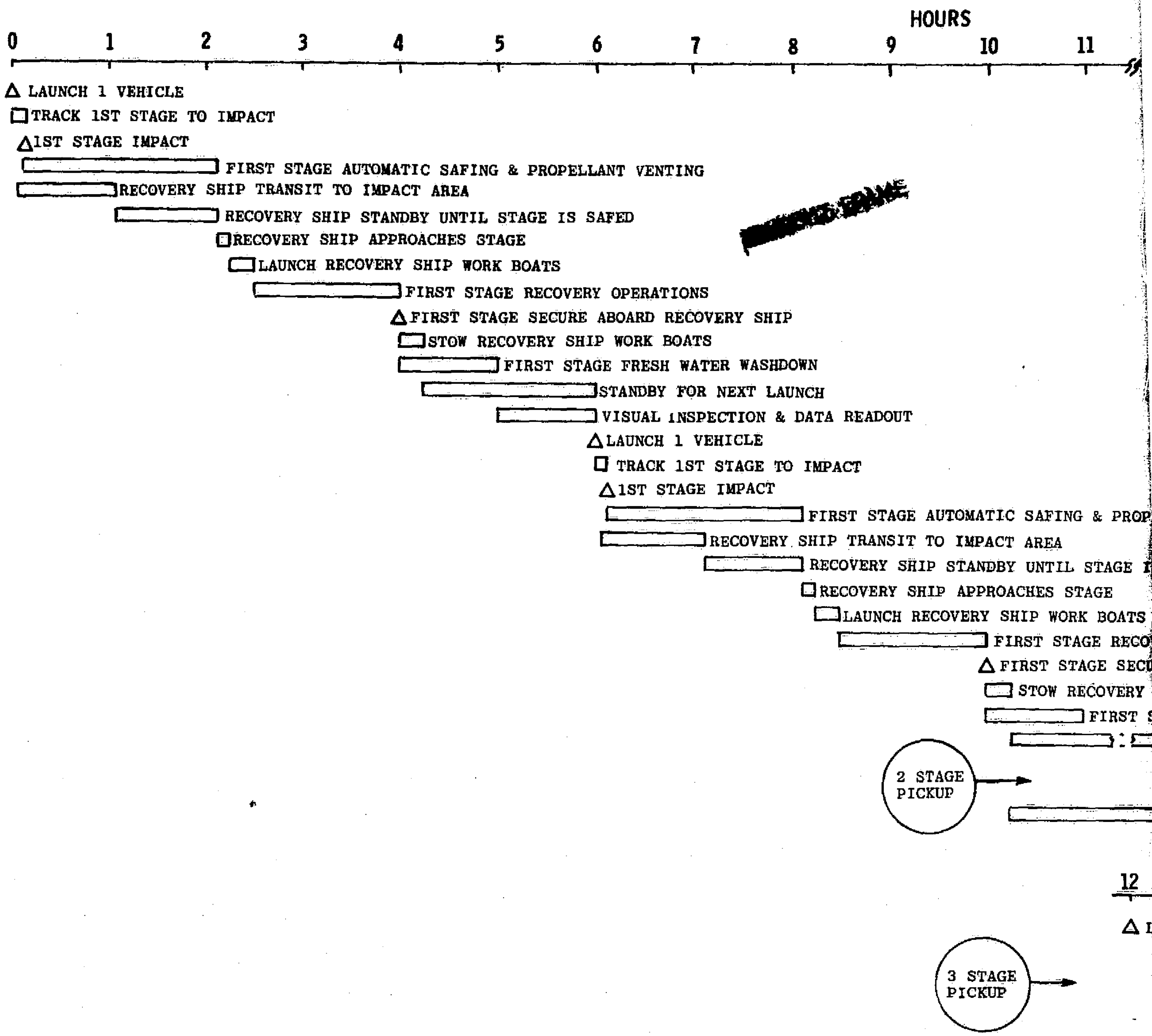
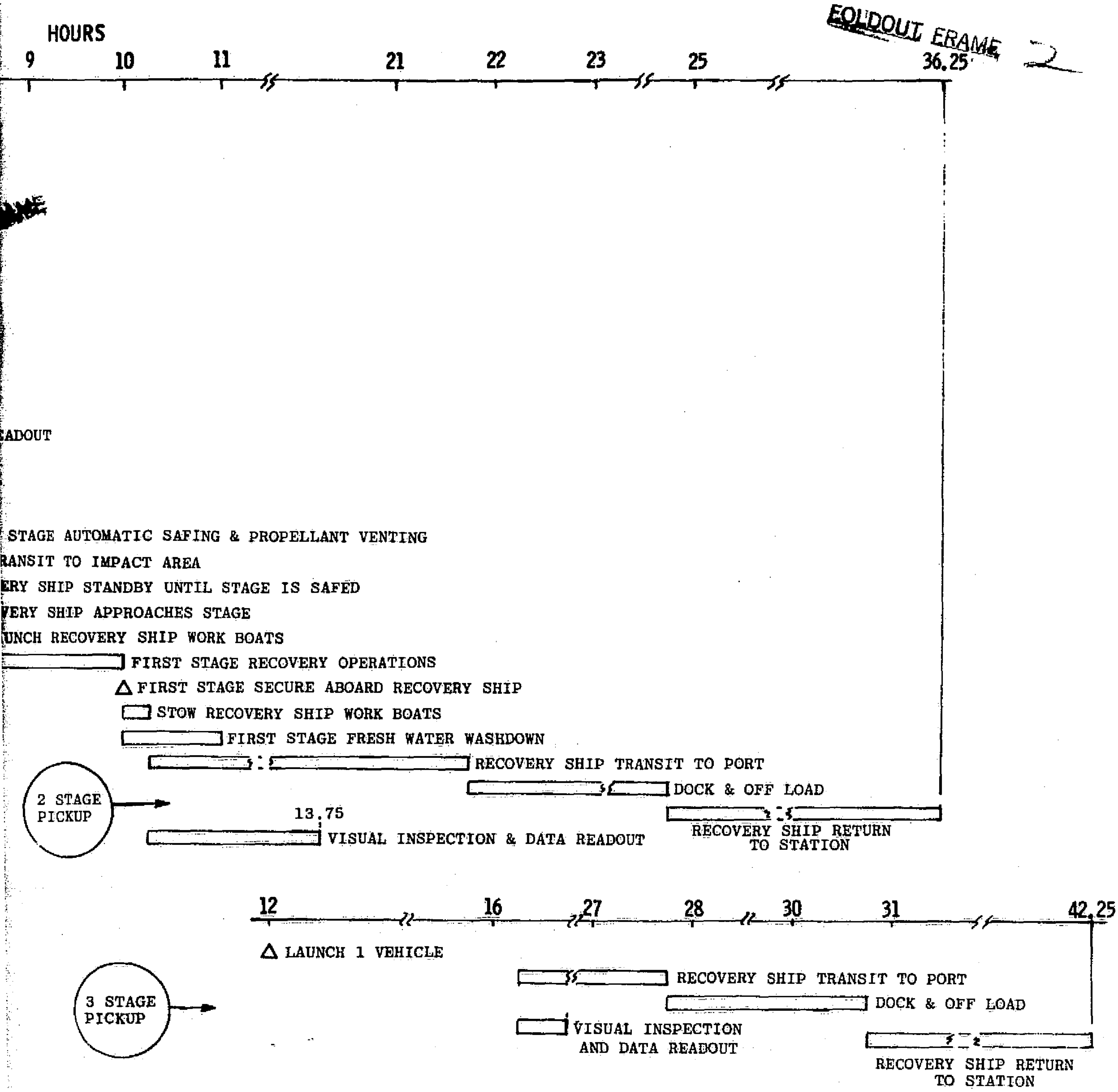


Figure 7.3-3. Stage 1 Recovery Oper



-3. Stage 1 Recovery Operations - Large HLLV

The recovery ship can approach Stage 2 sooner than Stage 1 because of the shorter propellant venting time. Parachute recovery operations are conducted by additional work boats concurrent with stage retrieval, but do add approximately 30 minutes to the serial time. After these stages and their parachute systems are retrieved, the ship returns to port while performing a fresh-water washdown and flight data readout of the stages. Recycle time of a Stage 2 recovery ship is 30-1/2 hours. The Stage 2 timeline is given in Figure 7.3-4.

7.3.5 STAGE PROCESSING

The large HLLV processing is essentially the same as the small HLLV with the exception of the timeline, Figure 7.3-5. Stage 1 requires 20 hours in processing due to the maintenance on 22 engines, while Stage 2 requires 14 hours. The stages are moved to the integration stations as required for buildup on the MLP.

7.3.6 PAYLOAD ACTIVITIES

The payload activities are the same as for the small HLLV, except for size.

7.3.7 TURNAROUND OPERATIONS

Large HLLV turnaround operations are shown in Figure 7.3-6. Stage 2 is the element with the longest turnaround time (128.5 hours). This is primarily due to its 24 hours on orbit prior to reentry. Stage 1 has a maximum turnaround time of 120.75 hours. Each recovery ship stays on station to recover three stages with the estimated Stage 1 recycle times of 108.75, 114.75, and 120.75 hours per shipload. The launch pads are capable of supporting a launch every 2.1 days, maintaing a launch rate of four per day with a normal crew support requirement of 24 hours per day, 5 days/week.

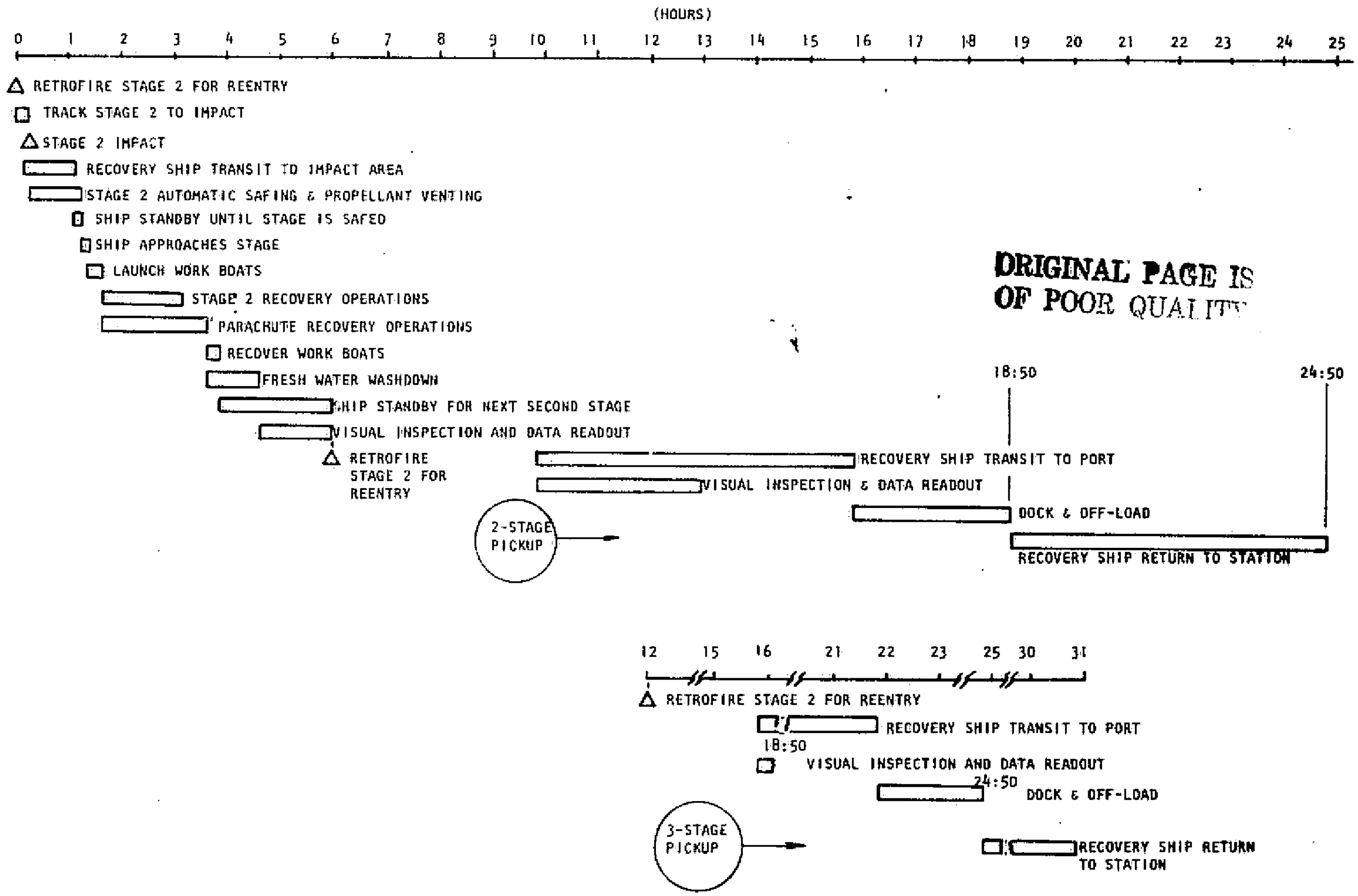


Figure 7.3-4. Stage 2 Recovery Operations - Large HLLV

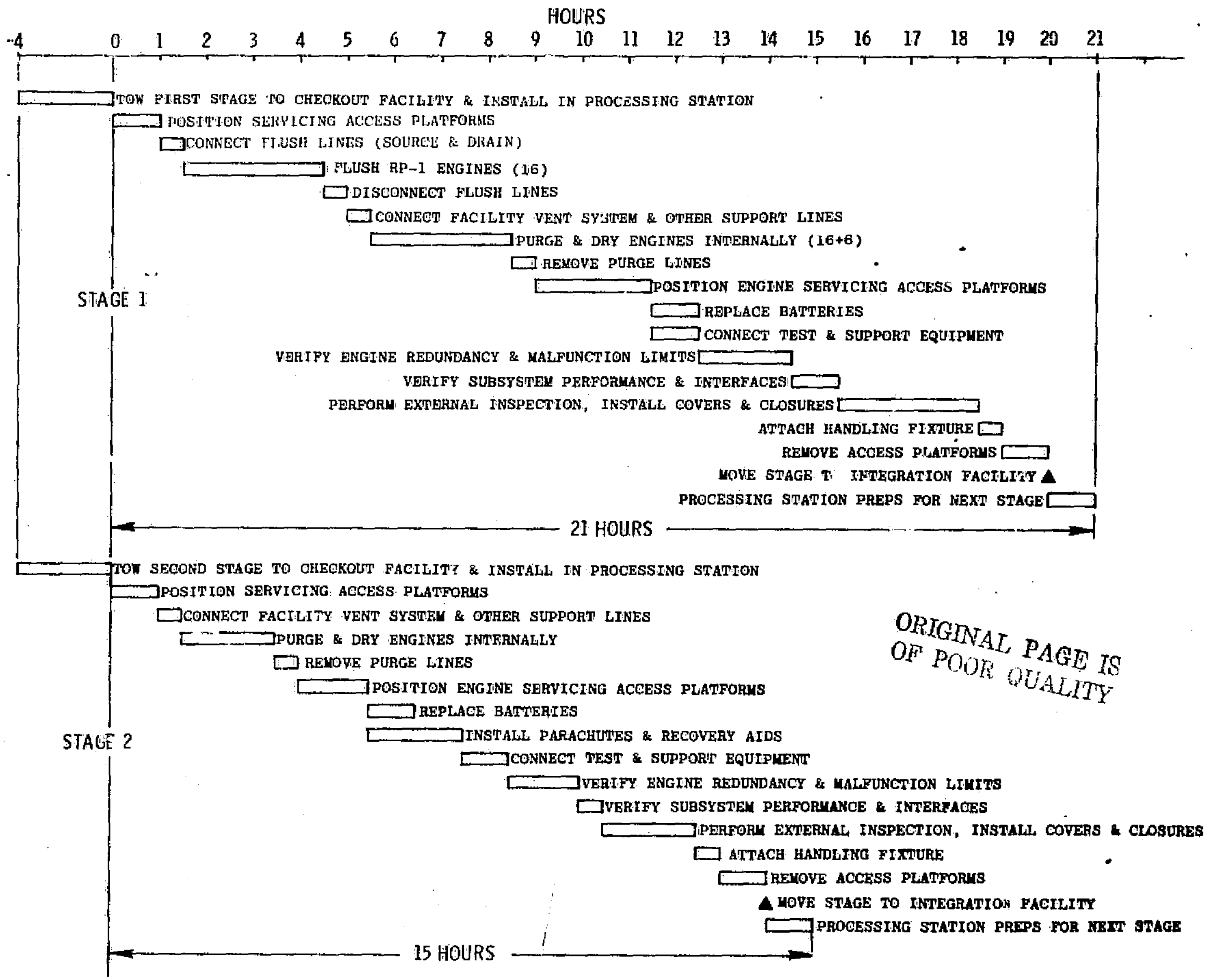


Figure 7.3-5. Stage Processing - Large HLLV

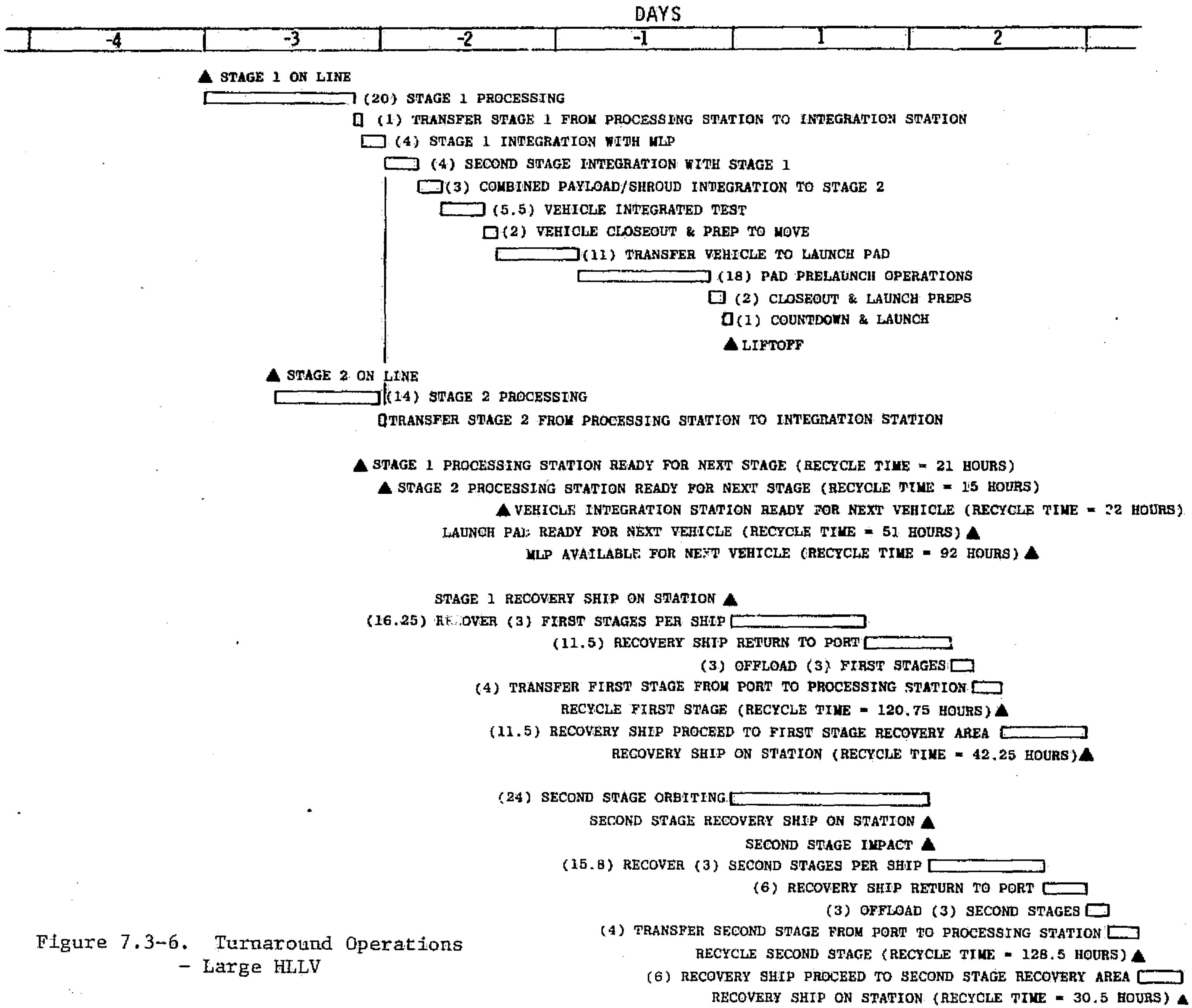


Figure 7.3-6. Turnaround Operations
- Large HLLV

7.4 BALLISTIC HLLV OPERATIONS/FACILITY REQUIREMENTS

The operational timelines provide the data necessary to estabilish facility, support equipment, and vehicle fleet size requirements to maintain the specified traffic model. These quantities are compared for the two ballistic HLLV's in Table 7.4-1. It is noted that the requirements to support the small HLLV represent a significant investment when compared to the large HLLV. Although the large HLLV represents a launch rate 25 percent of that of the small HLLV, the requirements range from 29 to 50 percent of those of the small HLLV. This is due to the extra servicing time for additional engines and the complexities of moving, servicing, and recovering the larger vehicle.

Table 7.4-1. HLLV Facility and Support Requirements

Item	Quantity Required *	
	Small HLLV	Large HLLV
Stage 1 processing station	14	6
Stage 2 processing station	12	4
Vehicle integration station	18	6
Launch site	33	10
Launch control center	33	10
Mobile launch platform (MLP)	57	17
MLP transporter	29	6
Stage 1 transporter	10	4
Stage 2 transporter	10	4
Stage 1 recovery ship	10	4
Stage 2 recovery ship	6	4
Port docking berth	4	2
Port unloading crane	8	4

*Totals exclude spares, service replacement (wearout), and periodic scheduled maintenance.

7.4.1 LAUNCH PAD SITING

An acoustic-level analysis was performed in order to site the launch pads, stage processing/vehicle integration buildings, and port facilities. A 30-dB limit was established between adjacent launch sites for maximum personnel exposure. A 20-dB limit was established between the launch pads and the processing/integration buildings, as well as between the launch pads and the port facilities, for maximum repetitive exposure to personnel. These limits are shown in Figure 7.4-1 for the expected acoustic levels from both HLLV configurations. Utilizing the requirements shown in Figure 7.4-1, and the 130/120-dB maximum pressure-level criteria, a typical facility siting was detailed in Figure 7.4-2 for each configuration HLLV.

When superimposed on a map of the KSC and adjoining communities, Figure 7.4-3, it may be seen that a launch complex of this magnitude would not only impact neighboring residential areas, but would also require the acquisition of some communities and placement of launch pads in ocean areas.

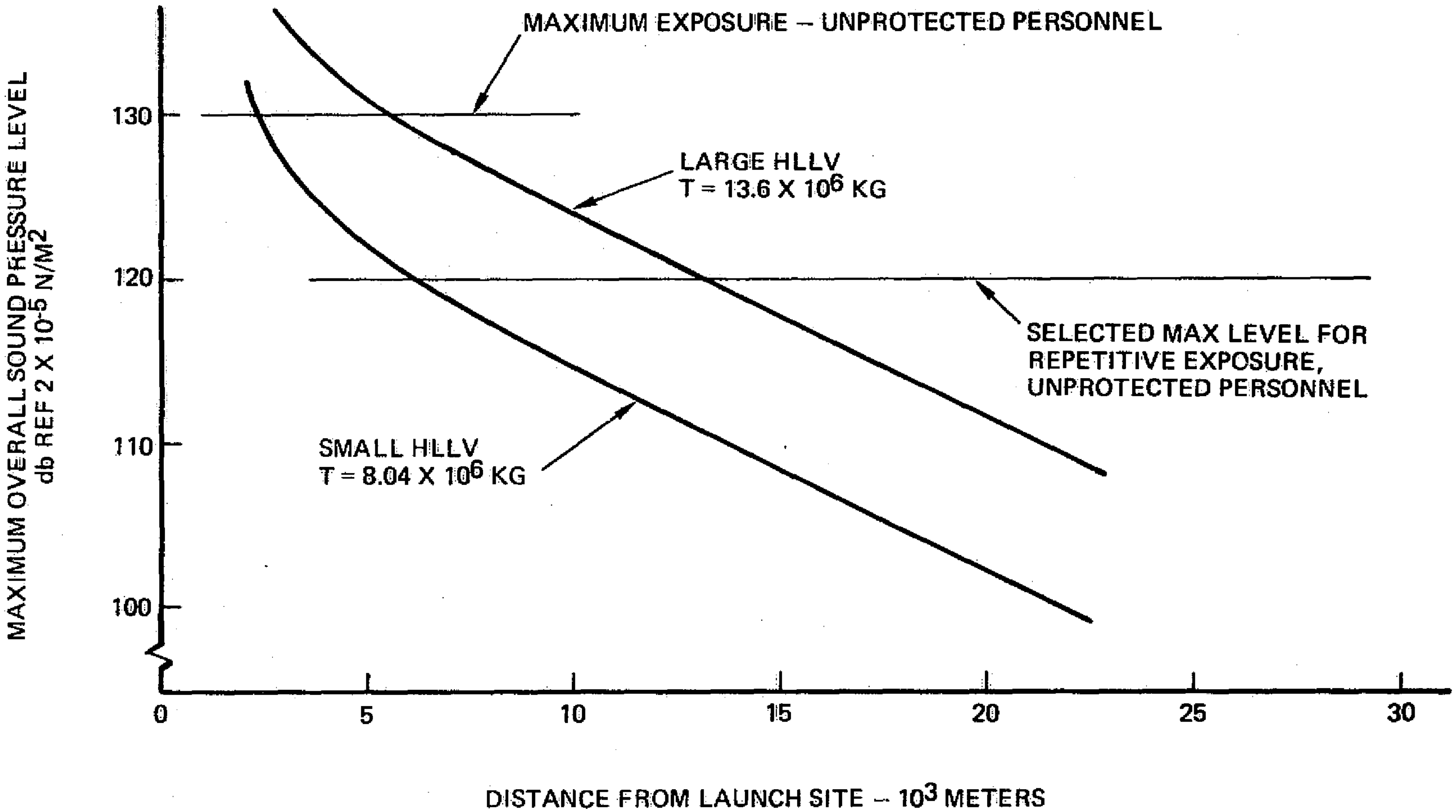


Figure 7.4-1. Personnel Acoustic Hazards

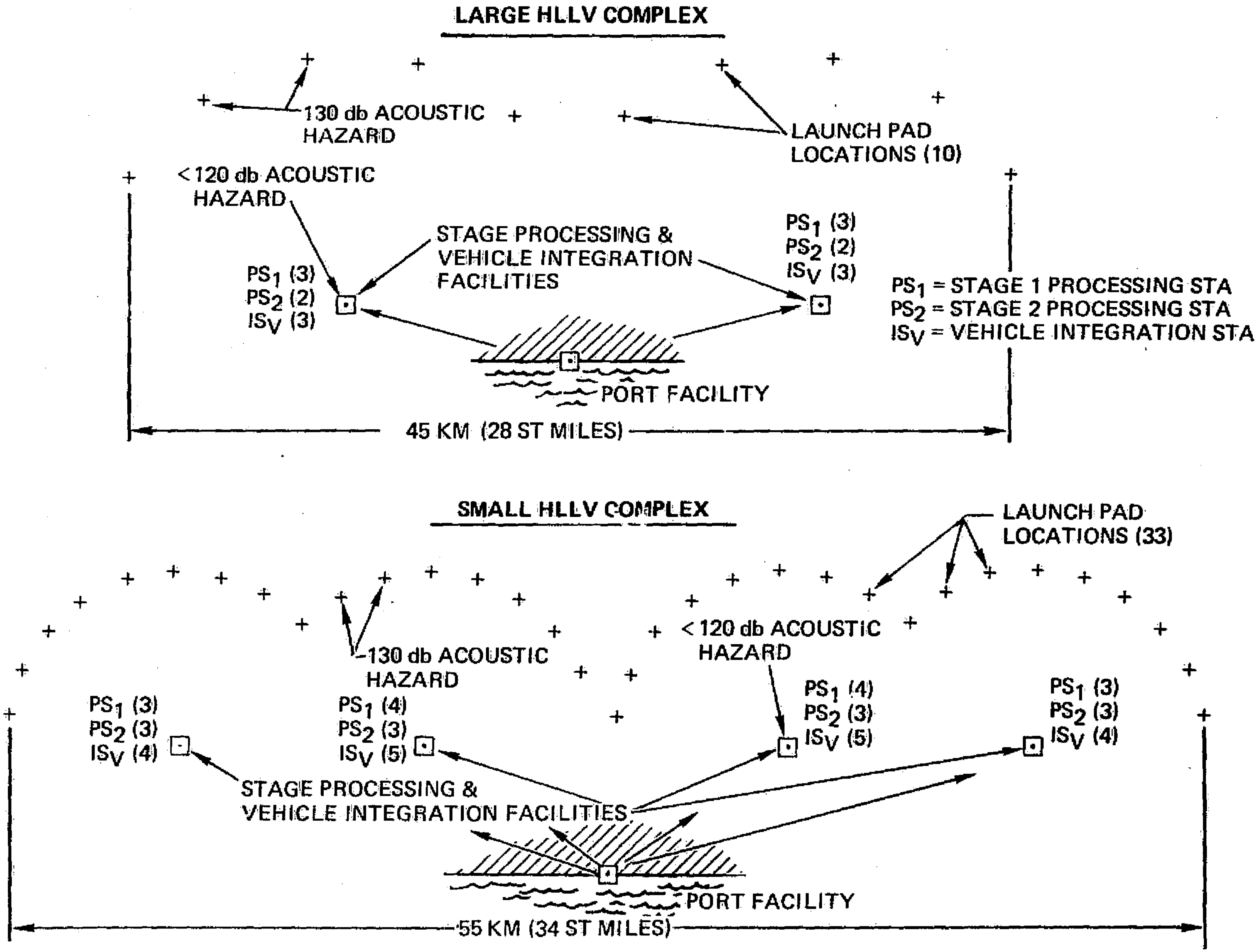


Figure 7.4-2. Typical HLLV Complex Layouts

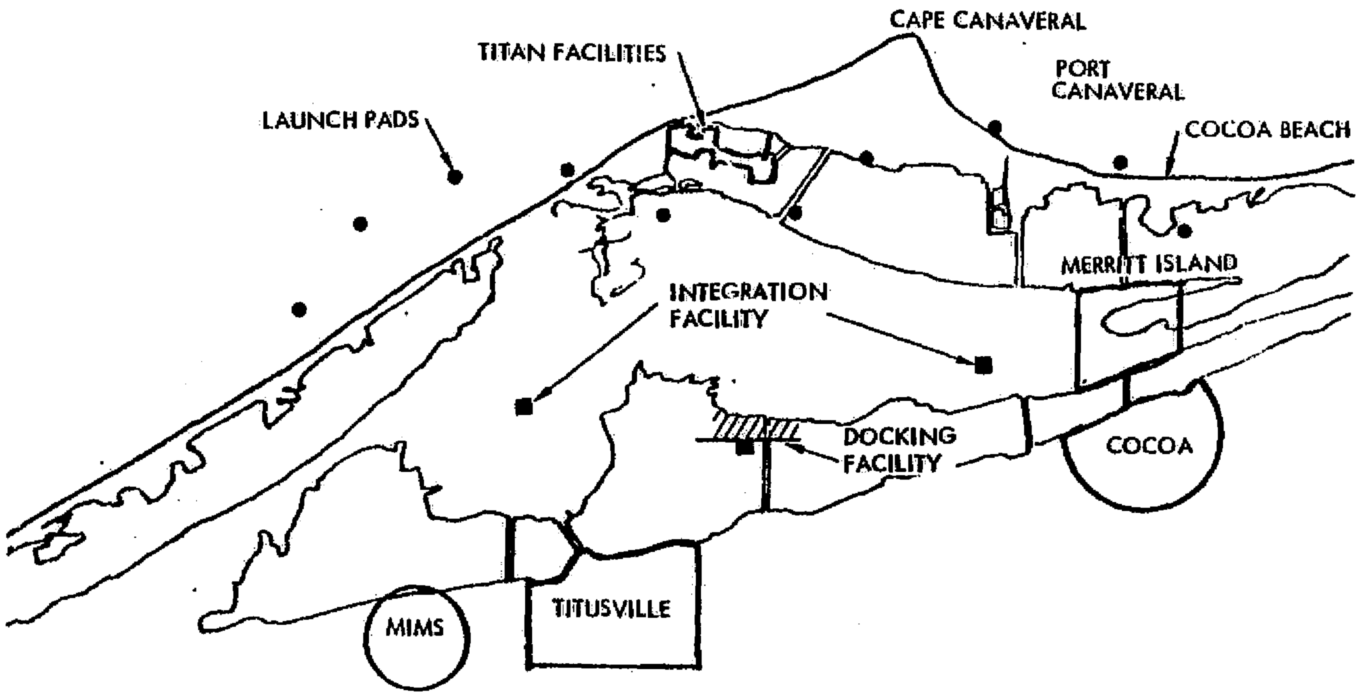


Figure 7.4-3. Large HLLV Launch Complex

7.4.2 PROCESSING/INTEGRATION FACILITY

Typical layouts for a combined stage processing/integration facility are shown in Figure 7.4-4 for the small HLLV, and in Figure 7.4-5 for the large HLLV. Combined facilities were chosen to minimize transit time between facilities. The small HLLV requires four facilities, while the large HLLV requirements are less than 50% of the small HLLV. The payloads arrive from their separate facility, ready for mating, and enter directly into the integration cycle. The crawler transporter enters the facility from the rear and takes the MLP/vehicle directly to the launch site.

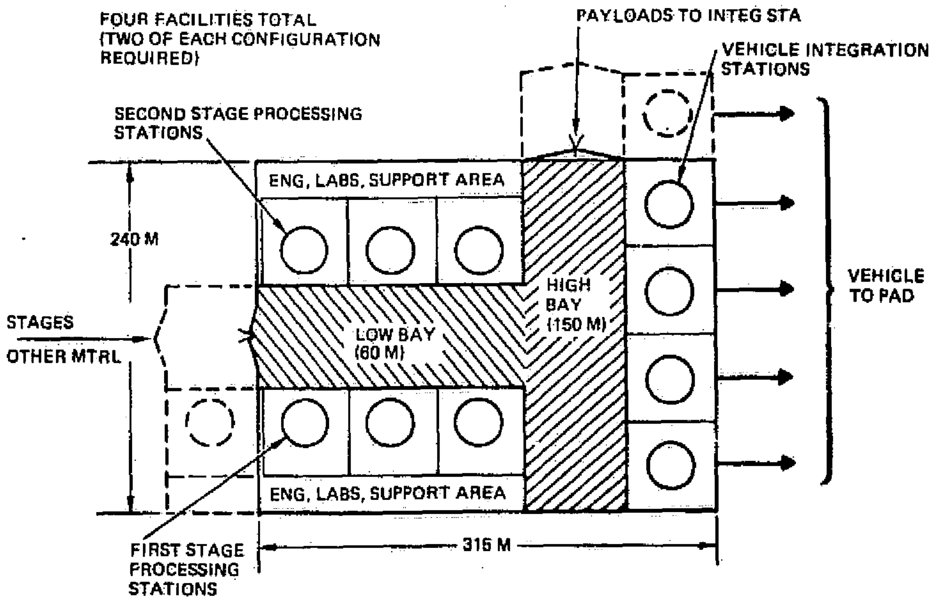


Figure 7.4-4. Stage Processing and Vehicle Integration Facility (Small HLLV)

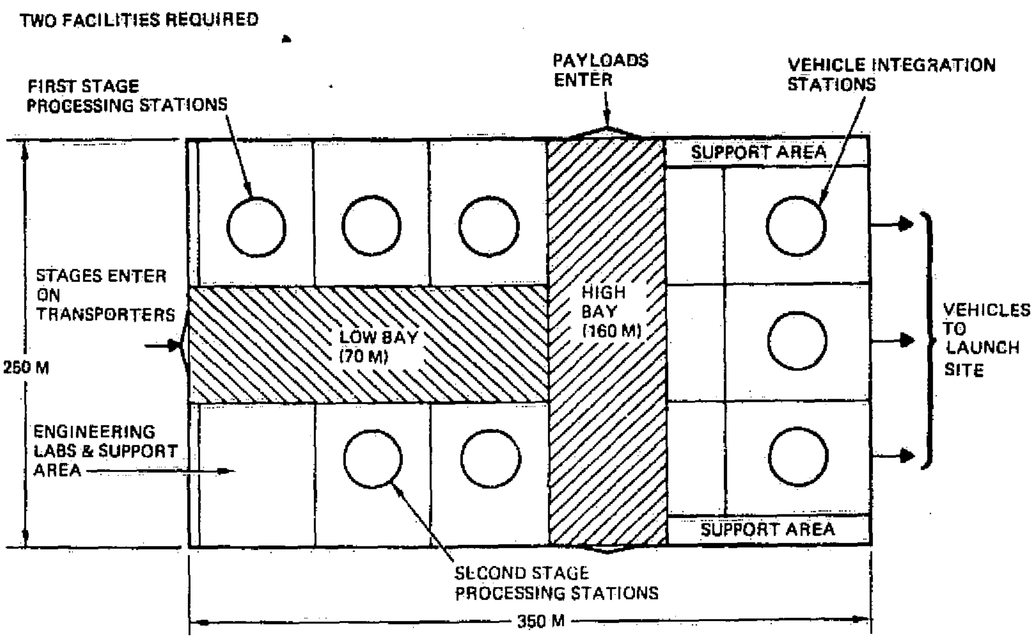


Figure 7.4-5. Stage Processing and Vehicle Integration Facility (Large HLLV)

7.4.3 ON-LINE VEHICLE REQUIREMENTS

The HLLV stage element requirements, shown in Table 7.4-2, are a combination of the number necessary to maintain the traffic model only. The issue of attrition rate was not addressed. The on-line requirements were calculated using the previously developed turnaround timelines. Engine replacement is scheduled following every twentieth flight. Based on planned Shuttle engine changeout times and combined leak/pressure check times, the additional stage requirements to accommodate engine changes range from 35 to 59 percent of the basic on-line requirements.

Table 7.4-2. HLLV Stage Element Requirements

REQUIREMENTS	SMALL HLLV		LARGE HLLV	
	STAGE 1	STAGE 2	STAGE 1	STAGE 2
ON LINE	61	70	22	23
SPARES (WEAROUT AND LOSS)	2	2	1	1
TOTAL ENGINE REPLACEMENTS	36	36	10	8
TOTAL	99	108	33	32

7.4.4 NATURAL ENVIRONMENT CONSIDERATIONS

Optimum Space Shuttle launch times relative to natural environment (NASA CR-150374, September 1977) was the source of data shown in Figure 7.4-6 for the probability of favorable launch conditions at KSC. These data are based on 14 years of weather data. It can be seen that the highest favorable launch probabilities exist from March through December, while the December through March time period offers the lowest probability of favorable launch conditions. Missed launches could then occur 65 to 78 working days/year. This high a number, although unlikely, might be made up on weekends or with additional daily launches if additional facilities are provided.

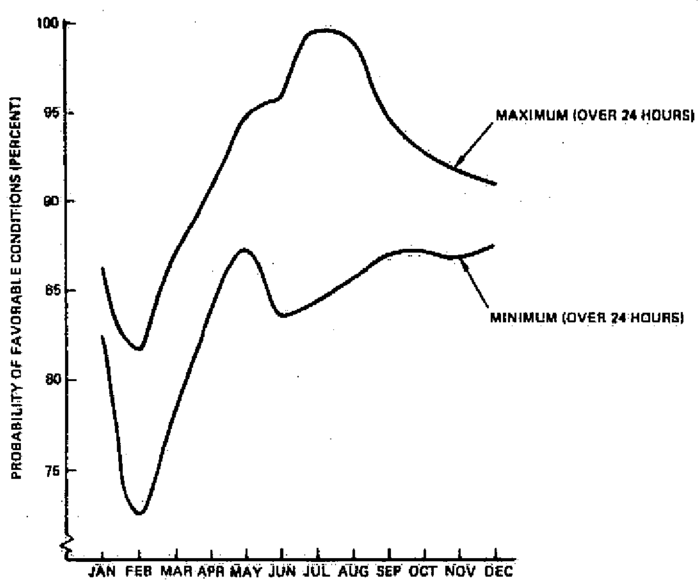


Figure 7.4-6. KSC Probability of Favorable Launch Conditions

Sea conditions in the recovery area are a factor which could also affect the launch/recovery options. For the vehicle stage sizes considered (27-m to 34-m diameter and 21-m to 31-m length), recovery should only be attempted in waves of less than 10 feet. This would restrict retrieval operations to a seal state of 4 or less. Table 7.4-3 shows the Atlantic Ocean sea conditions. The wave height where retrieval would be restricted is prevalent 15 percent of the time. This, again, will restrict launch/recovery operations or risk stage damage or loss during recovery in heavy seas.

Table 7.4-3. Atlantic Ocean Sea Conditions

WAVE HEIGHT (FT)	RELATIVE FREQ. (%)
0 - 3	20
3 - 4	25
4 - 7	25
7 - 12	15
12 - 20	5
20+	10

SEA STATE	WAVE DESCRIPTION	WIND VELOCITY (KNOTS)	MIN WIND DURATION (HR)	WAVE HEIGHT (FT)
3	SMALL	11-16	4	2-6
4	MODERATE	17-21	8	4-9
5	LARGE	22-27	10	6-16

REFERENCE: Handbook of Oceanographic Tables (1966)

7.5 WINGED HLLV OPERATIONS

The total turnaround operational flow for a winged HLLV is shown in Figure 7.5-1. Flight-readiness was chosen as the beginning of a cycle. After completing flight-readiness, which is comparable to that performed on a large jet (747 type), the vehicle takes off on turbofan power. At approximately 1000-ft altitude, the takeoff gear is jettisoned. The gear lands by parachute, is recovered, and returned to its refurbishment facility. The winged HLLV continues to the equator where, after an airbreathing engine powered climb, injection into orbit is accomplished with SSME-type rocket engines. Once in orbit, the crew compartment is swung aside and cargo unloaded. Down-cargo is then loaded if required. The crew compartment is repositioned and secured for reentry. The vehicle reenters, flies back utilizing its airbreathing engines, and lands at its launch site.

The vehicle taxies to the propellant off-load area, the crew egresses, and the vehicle is prepared for off-loading and inerting of the LH_2 system. Hypergolic systems will periodically be drained and flushed. After completion of safing, the vehicle is towed to the maintenance facility where it is jacked

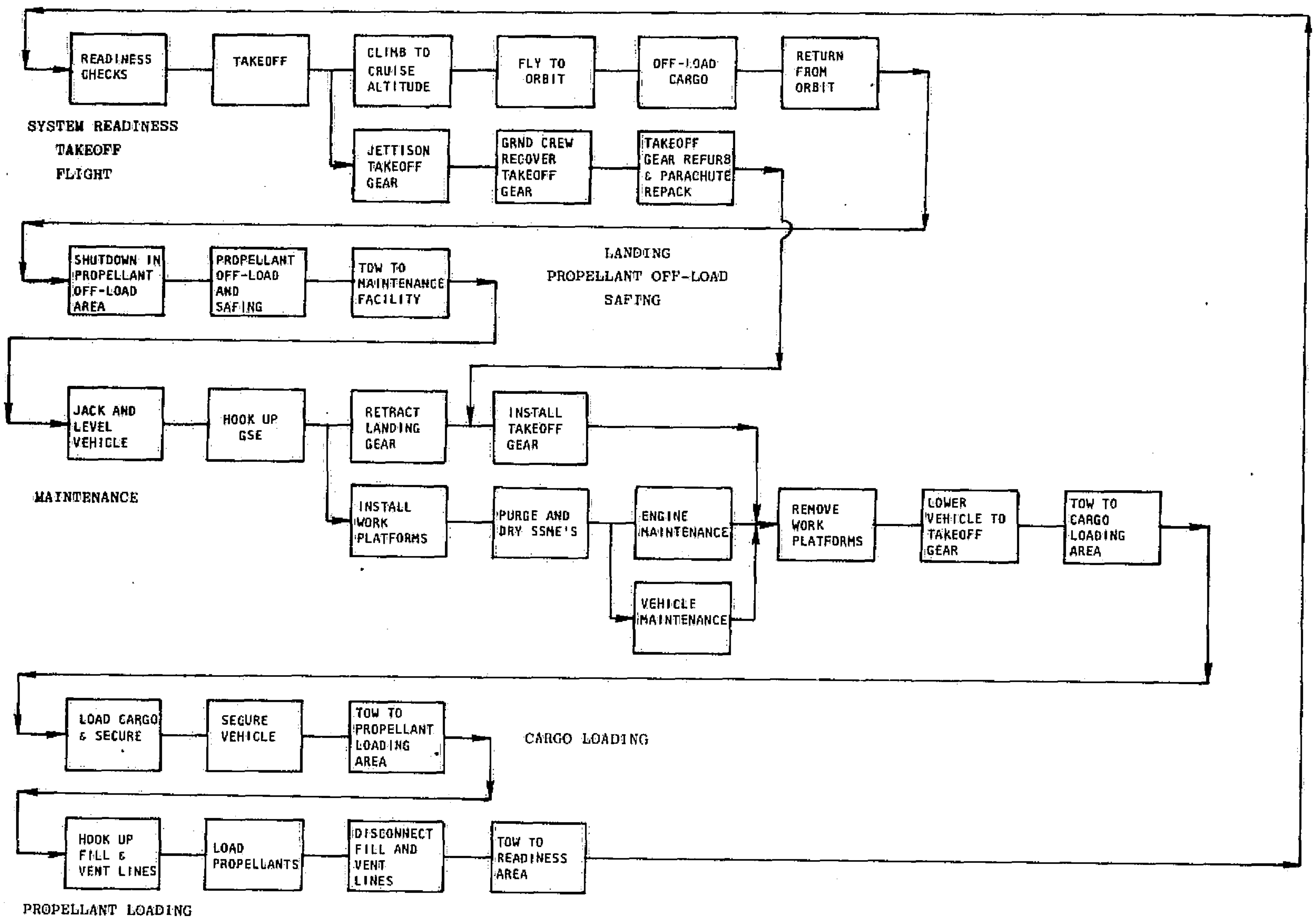


Figure 7.5-1. Operational Flow - Winged HLLV

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and leveled and, after retraction of the landing gear, prepared for installation of the takeoff gear. The SSME-type engines are purged and dried, and engine/vehicle maintenance performed. The vehicle is then towed to the cargo loading area. After cargo loading, the vehicle is towed to the propellant loading area for main and auxiliary propulsion system loading. After completion of loading, the vehicle LO_2/LH_2 topping and venting are maintained with a mobile unit while the vehicle is towed to the readiness area.

7.5.1 WINGED HLLV TIMELINE ANALYSIS

A timeline analysis of the various operations was made to determine turnaround times for the purpose of establishing facility, equipment, and fleet size requirements.

7.5.2 FLIGHT OPERATIONS

Flight operations encompass three major segments of the total turnaround: system-readiness checks and takeoff, flight, and landing/safing (Figure 7.5-2). Flight time to the equator and into orbit takes approximately three hours. Once in orbit, 15 minutes are allocated to orbit adjustment with the OMS and stabilization with the RCS. The vehicle is unloaded within two hours; then reenters and returns to its launch site in three hours. Following cryogenic off-loading, the vehicle is considered safed and is towed to the maintenance facility for normal turnaround servicing. The flight operations segment requires 12.5 hours.

7.5.3 LAUNCH OPERATIONS

Launch operations are divided into three main areas of vehicle servicing: maintenance, cargo loading, and propellant loading, as illustrated in Figure 7.5-3. At the maintenance facility, the takeoff gear and its parachute system are reinstalled and verified, work platforms installed, SSME's purged and dried, and routine engine and vehicle servicing performed. The vehicle is in the maintenance facility 11.5 hours. The vehicle is then towed to the cargo loading area where the payload (6×6×30 m, palletized) is installed. This operation is accomplished in two hours. The vehicle is then towed to the propellant loading area; here, the auxiliary and main propulsion system propellants are loaded. Propellant loading is the longest of the launch operations (15.75 hours). Total ground operations time for the winged HLLV is 30.75 hours. Table 7.5-1 shows the propellant quantities required by system. Flow rates of 25 GPM were chosen for the auxiliary systems because of loading accuracy requirements. The LO_2 flow rate of 1000 GPM was modified from the Shuttle LO_2 load rate. The LH_2 flow rate to the wing tanks (500 GPM) is comparable with fuel fill rates of large jets such as the C-5A and 747. This rate is through six feed lines for a total vehicle loading rate of 3000 GPM. Should the vehicle integrity and ground loading system prove capable of higher rates, the ground turnaround of the vehicle could be improved significantly.

7.5.4 TURNAROUND OPERATIONS

The turnaround times for all the main elements of the winged HLLV system are shown in Figure 7.5-4. The winged HLLV requires 43.25 hours/cycle. Almost

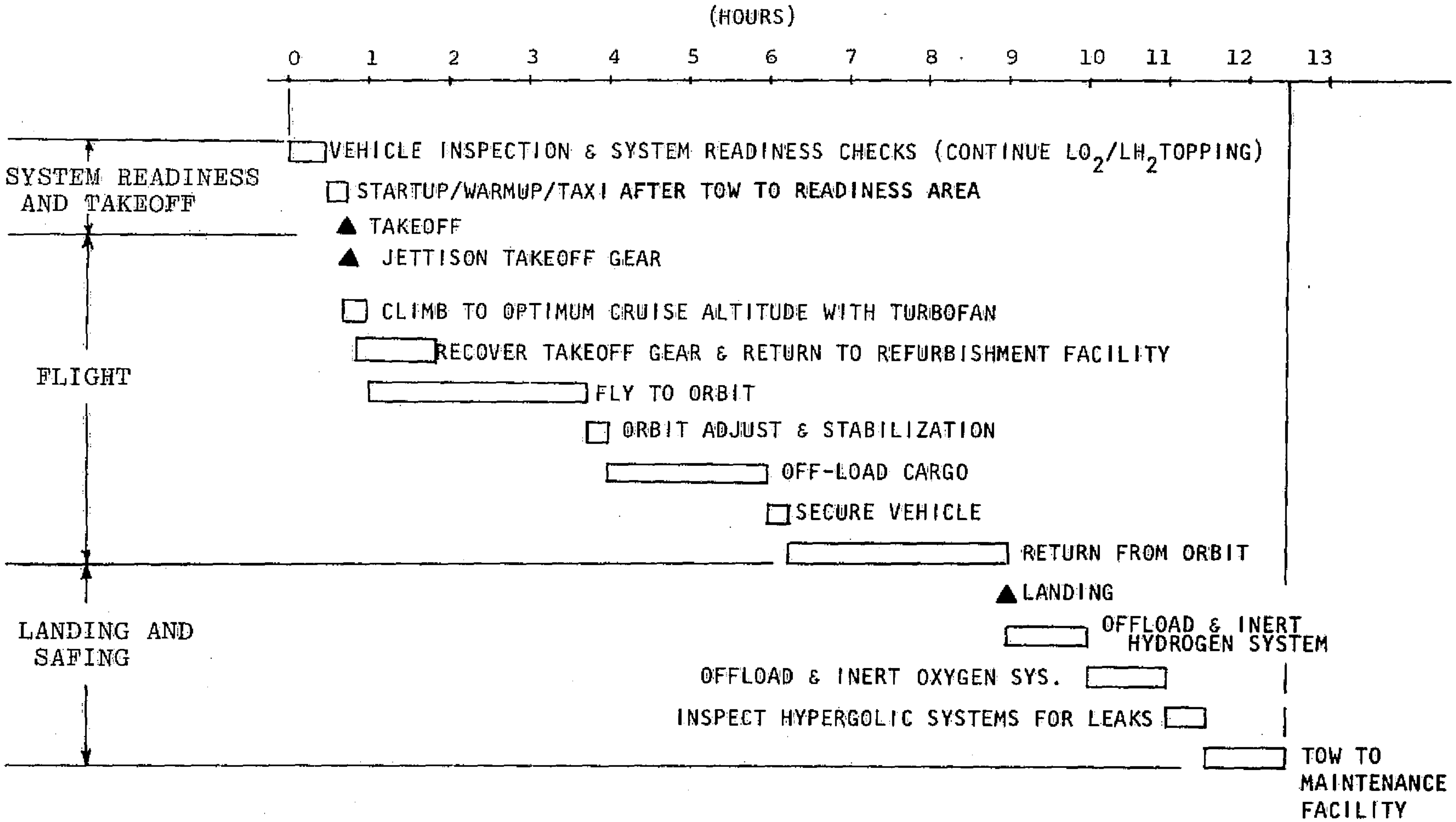

Figure 7.5-2. Winged HLLV Flight Operations

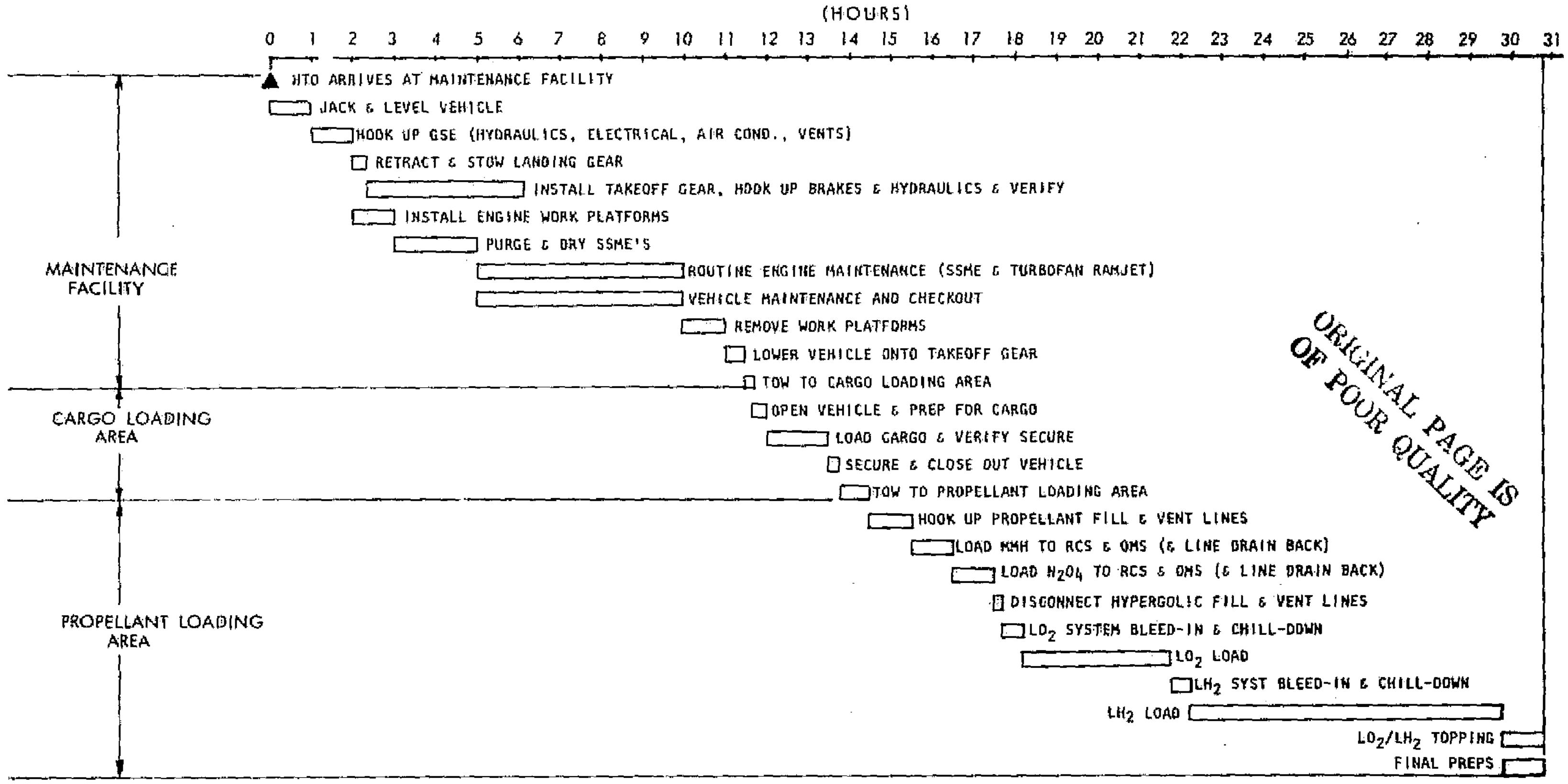

Figure 7.5-3. Winged HLLV Launch Operations

Table 7.5-1. Winged HLLV Propellant Requirements and Load Rates

SYSTEM	PROPELLANT	QUANTITY (LB)	FLOW RATE (GPM)	LOAD TIME (HR)
RCS	N_2O_4 MMH	4,250 2,660	25 25	0.25 0.25
OMS	N_2O_4 MMH	11,270 7,040	25 25	0.60 0.65
MAIN PROPUL-SION	LO_2 LH_2	2,260,200 893,800	1000 500*	4.0 8.0
TOTAL	--	3,179,220	--	13.75
*TO SIX TANKS				

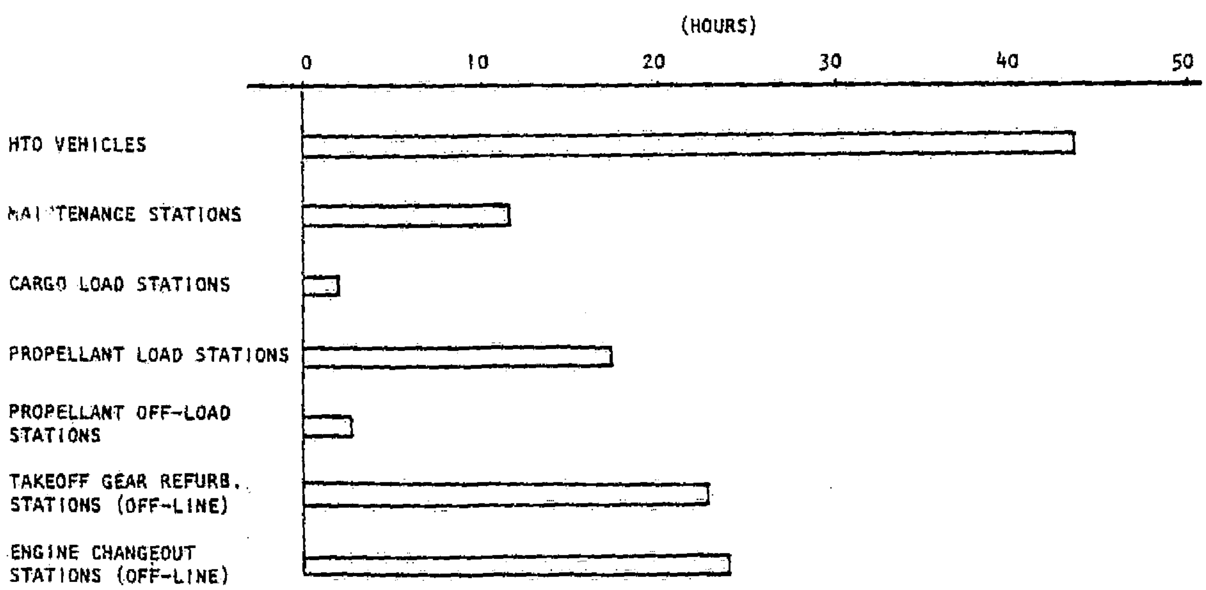

Figure 7.5-4. Winged HLLV Turnaround Operations

Almost one third of this time is taken up in the propellant loading facility. The maintenance time of 10.75 hours appears long for a typical aircraft servicing. However, SSME servicing, which is comparable to the Space Shuttle program, accounts for a large segment of this time. Once actual Space Shuttle data are accumulated on SSME engines, service time may be reduced.

7.6 WINGED HLLV OPERATIONS/FACILITY REQUIREMENTS

A combination of all the operations timelines resulted in the element turnaround times. These times and the traffic model requirement for 16 flights/day enabled an estimation of facility and vehicle quantity requirements, as shown in Table 7.6-1. The maintenance, cargo storage, takeoff gear refurbishment, and engine changeout facilities are the only structural buildings required. The cargo loading area is simply a winged HLLV parking ramp near the cargo storage facility. The propellant off-load and load facilities are open areas with access platforms, propellant lines, propellant loading units, storage tanks, and vent towers.

Table 7.6-1. Winged HLLV Quantity Requirements

HTO VEHICLES	30
MAINTENANCE STATIONS	10
CARGO LOAD STATIONS	3
PROPELLANT LOAD STATIONS	12
PROPELLANT OFF-LOAD STATIONS	4
TAKEOFF GEAR REFURB STATIONS	16
ENGINE CHANGEOUT STATIONS	4
RUNWAY	1

Winged HLLV operations are best conducted in an isolated area (remote from populated areas). This will eliminate potential environmental problems associated with inhabited areas. Figure 7.6-1 represents a typical winged HLLV facility layout. The spacing of propellant areas to each other and to the non-hazardous service areas could encompass a distance of 8 km. This would enable each propellant area to operate independently of the other.

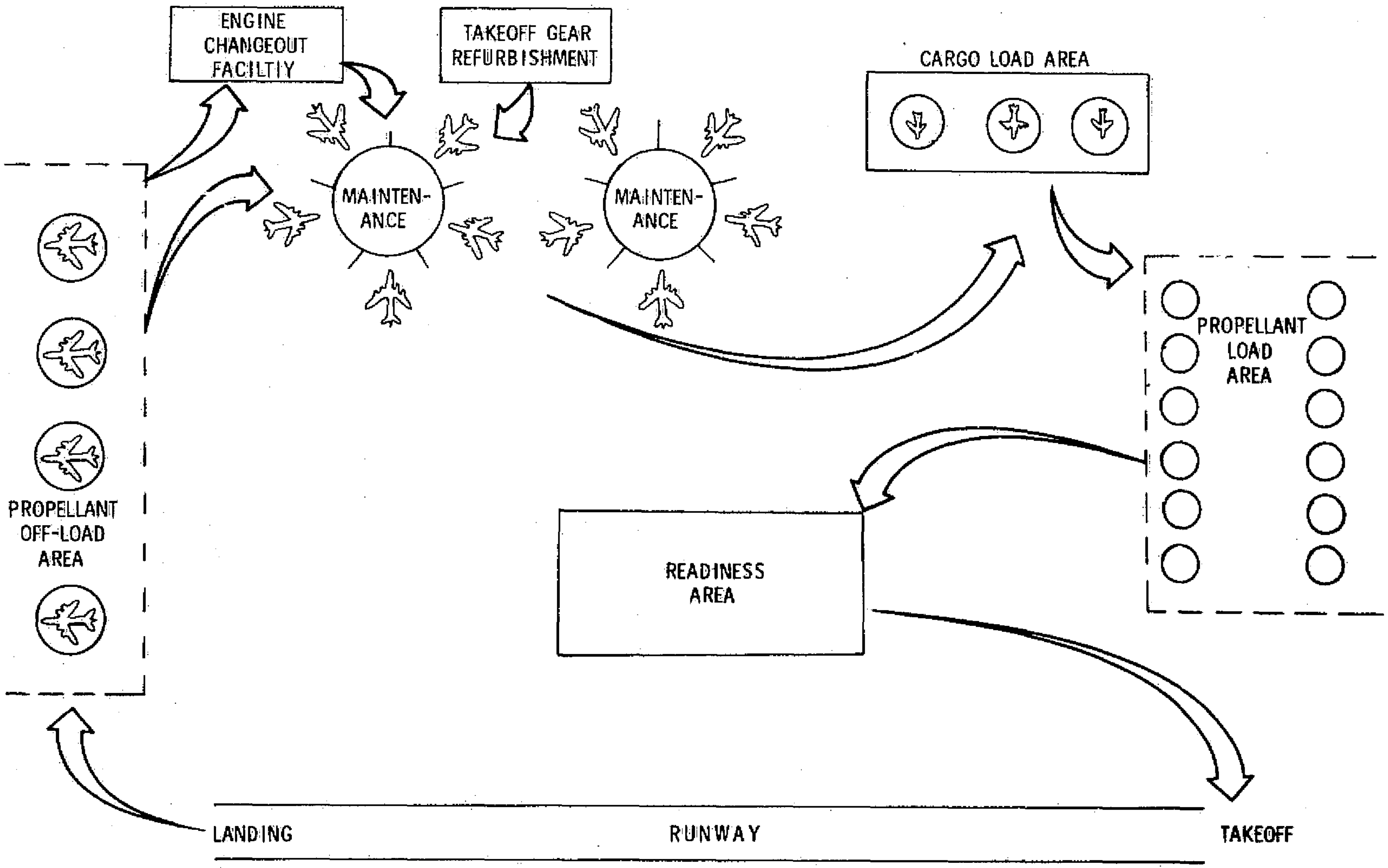

Figure 7.6-1. Winged HLLV Facility Layout

7.7 HLLV OPERATIONS COMPARISON

Some of the key areas of comparison between the two-stage ballistic HLLV and the winged single-stage-to-orbit configurations are presented in Table 7.7-1. Because of the larger payload capability of the ballistic HLLV, four launches/day are required as opposed to the 16 winged vehicle flights. In order to meet the launch rate requirement, 10 launch pads are required for the ballistic HLLV, whereas a single runway may be employed for the winged vehicle. Because of the stacking requirement for the two-stage ballistic HLLV, two high bay vertical assembly buildings are required as opposed to two aircraft maintenance-type buildings for the winged vehicle. In addition, considerable heavy handling equipment is required for moving and stacking of the ballistic HLLV. The difference in the number of processing stations is due to two-stage vs. single-stage processing. The turnaround time for the winged vehicle is approximately one-third that for the ballistic HLLV, primarily because of the recovery method and no requirement for mating and stacking. The risk of recovery damage from water impact is considerably higher than that for aircraft landing. The launch site area requirements are driven by the number of launch pads required and the necessary separation required to minimize the acoustic hazard for personnel/equipment. The maximum allowable level without ear protection is 130 dB, and 120 dB is the maximum allowable on a repetitive basis.

Table 7.7-1. HLLV Operations Comparison

	TWO-STAGE BALLISTIC	HORIZONTAL TAKEOFF
• LAUNCH RATE/DAY	4	16
• LAUNCH WINDOWS/DAY	2/ORBIT (3 HR)	12/ORBIT (CONT.)
• LAUNCH PADS	10	1
• FACILITIES	TWO HIGH-BAY BUILDINGS	2 A/C MAINT. TYPE
• HANDLING	CRANES, TRANSPORTERS, TUGS, MOBILE LAUNCH PLATFORMS AND CRAWLERS, AND RECOVERY SHIPS	TOW VEHICLE
• PROCESSING STATIONS	16	10
• ENGINES/VEHICLE	24	11-14
• FLEET SIZE (W/O ATTRITION)	22 FIRST STAGES 23 SECOND STAGES	30
• TURNAROUND TIME (DAYS)	5.5	1.8
• LAUNCH PAD REFURB.	EXTENSIVE	NIL
• RISK OF RECOVERY DAMAGE	HIGH	NIL
• ACOUSTIC LEVELS	130 dB @ 5.6 km 120 dB @ 13 km	<120 dB @ 1 km
• LAUNCH SITE AREA REQUIRED	850 km^2	< 20 km^2

8.0 PREFERRED TRANSPORTATION SYSTEM CONCEPTS

8.0 PREFERRED TRANSPORTATION SYSTEM CONCEPTS

Based upon the SPS and transportation system studies conducted to date, preferred transportation system characteristics, approaches and/or concepts have been identified. Although the depth of analyses to date have not been sufficient to lead to a specific transportation system selection, the results do indicate a direction in which further studies should be pursued.

8.1 EARTH-TO-LEO TRANSPORTATION

The preferred approach is a horizontal takeoff and landing HLLV with a payload capability in the order of 100,000 kg. The primary advantages of this concept are:

1. The ability to achieve the required launch rates and the launch site flexibility required to maintain those launch rates.
2. The method of recovery minimizes risk of damage and enhances turnaround time.
3. Minimum facility, equipment and ground operations requirements.
4. Environmentally acceptable acoustic level emissions.
5. Resolves the issue of space debris accumulation by virtue of its' inherent down payload capability.

8.2 LEO TO GEO TRANSPORTATION

The preferred COTV approach is a dedicated electric argon ion OTV. The primary advantages of this concept are:

1. Does not have the potential nuclear contamination concerns of a GCR and avoids the issue of sociopolitic aversion to nuclear systems in general.
2. Reduces the total mass to orbit requirements and consequently the HLLV flight requirements.
3. Avoids the SPS penalties associated with the self-propelled SPS approach.
4. Provides a higher reliability and less risk of damage during orbital transfer.

A common stage chemical (LOX/LH_2) POTV is preferred over the nuclear GCR primarily because of the potential nuclear contamination hazard and sociopolitic resistance to nuclear systems.

9.0 END-TO-END ANALYSIS

9.0 END-TO-END ANALYSIS

9.1 ANALYSIS OVERVIEW

The end-to-end analysis of the selected satellite (Figure 9.4-2) and rectenna concept is directed towards quantitive definition of key operations associated with major system elements and functions throughout the initial 30 years of the SPS program. The logic flow of this analysis is depicted in Figure 9.1-1 and begins with the SPS point design as defined in Volume IV of this document. Within each box, the major design concepts, functional analysis, and systems, facilities, mass flows and operational requirements that are considered in the analysis are listed.

The analysis first requires definition of the construction, operational, and maintenance concepts for both the satellite and the rectenna. From these the time-phased mass flows on earth and in space are derived. Definition of the space transportation system provides the constraints for development of the cargo packaging concept which, together with the satellite production rate and construction concept, provide the basis for development and/or definition of other system elements and operations. These include development of the space traffic model; definition of operations and facilities at LEO; and definition of the launch complex operations, mass flows and facilities.

Similarly the rectenna site construction, operations, mass flows and facilities are established. The earth manufacturing requirements, the location of manufacturers and earth logistics concept are then defined to provide best support of the satellite and rectenna construction and maintenance requirements.

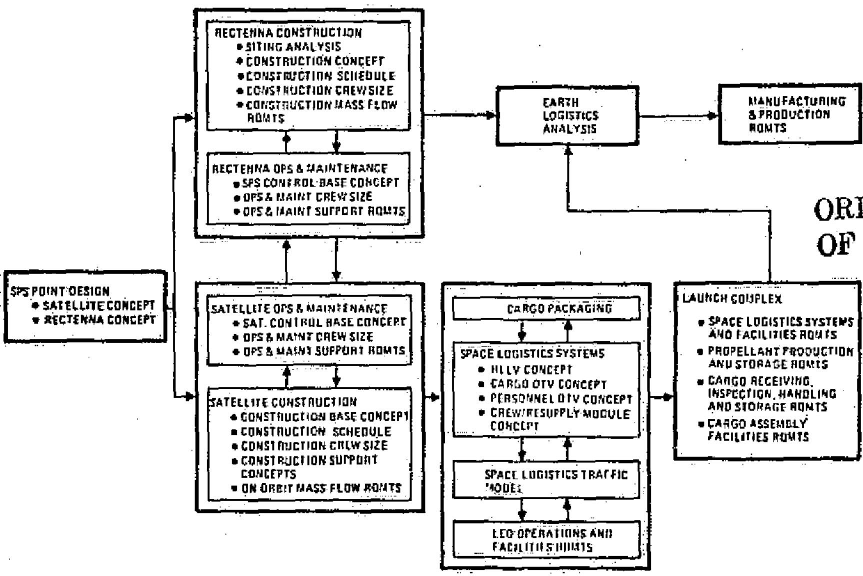

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Figure 9.1-1. Major Elements of the End-To-End Analysis

9.2 SPACE OPERATIONS

The operations concept centers around three areas of activity: launch sit., LEO, and GEO as pictured in Figure 9.2-1. After establishment of the LEO and GEO bases, the program is initiated by HLLV flights which transport satellite construction material to LEO for transfer to COTV's and delivery to GEO. The SPS construction crew is carried to LEO by HLLV's and utilize POTV's for transit to GEO, returning in the same vehicles when relieved. These operations are discussed in more detail below.

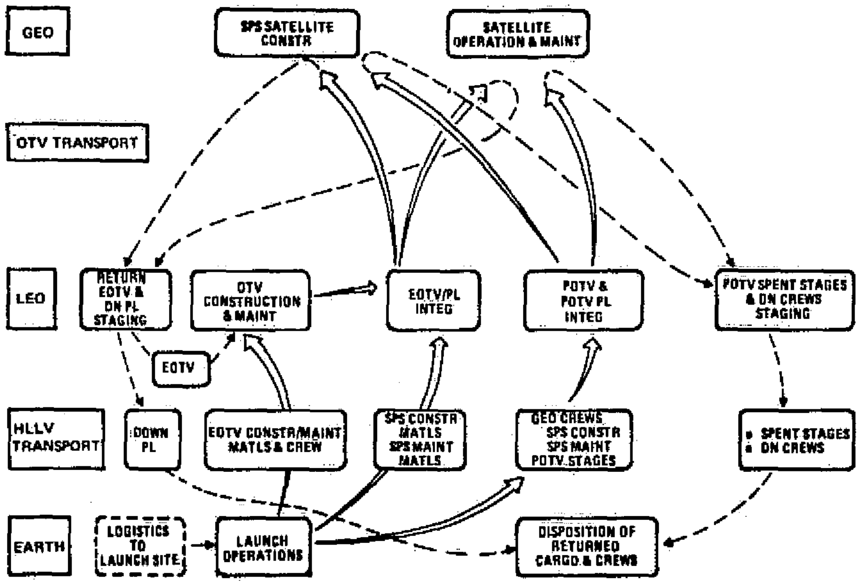

Figure 9.2-1. Space Operations Concept

Launch site facilities and operations are dictated by the mass flow of material, equipment and personnel which must be transported to orbit. A summary of mass flows to orbit requirements for both satellite construction and operational maintenance over the 30 year program period is given in Table 9.2-1. Table 9.2-1 is based on a satellite mass of 38.04×10^6 kg, including growth and packaging allowances. (Although the weight statement for the point design satellite varies from time to time, the mass baseline for the table is currently representative.) Not included in the table is the approximately 17×10^6 kg required for construction of each set of COTV's, eventually totaling 4½ sets of 10 vehicles per set. (A COTV set is defined as the number of vehicles required to transport the construction mass for one satellite.) Orbital operations analysis accomplished to date have been largely concerned with development of a viable satellite construction concept during the mature portion of the program, which is subsequent to establishment of the base facilities in LEO and GEO. Concepts for these bases are described in Sections 9.3 and 9.4.1 and will be developed in greater detail during the follow-on study; at that time the number of COTV's which will be required to support construction of the GEO facility will be determined.

Table 9.2-1. Space Construction and Operations Scenario

CONSTRUCTION AND MAINTENANCE MASS															PERSONNEL MASS								SUMMARY			
		SATELLITE CONST. SCHEDULE											HLLV FLIGHTS (NOTE 3)		SATELLITE CONSTRUCTION CREW				MAINTENANCE CREW							
PROGRAM YEAR	CALENDAR YEAR	PER YEAR	CUM TOTAL	CONSTRUCTION TIME PER SATELLITE (DAYS)	COTV-DELIVERED SAT. - CONSTR.-RELATED MASS (NOTE 1)	TOTAL CONSTR. MASS TO GEO EA. YR (INCLUDES 30% GROWTH)	SATELLITES OPNL AT YEAR END	NO. COMPLETED SAT. YEARS AT YEAR END	COTV-DELIVERED SAT. MAINT. MASS DURING YR (NOTE 2)	TOTAL YEARLY MASS TO GEO (CONST. + MAINT.)	COTV DELIVERY REQMT PER YEAR (CONST. + MAINT. MASS + 10% PKG MASS)	COTV FLIGHTS (NOTE 3)	SCHED. COTV P/L MASS	HLLV FLIGHTS	CREW ROTATIONS EACH 90 DAYS (68 MEN)	POTV FLIGHTS (NOTE 4)	POTV DELIVERED CREW + CONSUMABLES MASS (NOTE 4)	HLLV FLIGHTS (NOTE 3)	CREW ROTATIONS @ 4 CREWS PER SATELLITE YEAR (24-MEN CREWS)	POTV FLIGHTS @ 0.5 FLTS/CREW ROTATION LEO TO GEO	POTV DELIVERED MASS (NOTE 5)	HLLV FLIGHTS (NOTE 3)	TOTAL MASS (10^6 KG)	TOTAL POTV FLIGHTS	TOTAL COTV FLIGHTS	TOTAL HLLV FLIGHTS
	1996	-																								
	1997	-																								
1	1998	1	1	180		34.11	1	0.5	0.46	34.58	38.04	10	39.4	433	2	28	1.06	84	2	1	0.04	3	39.14	29	10	520
2	1999	2	2, 3	180		68.22	3	1.5	1.396	69.62	76.58	19	74.9	823	4	56	2.12	168	6	3	0.11	9	78.81	59	19	1,000
3	2000	3	4, 5, 6	120		102.33	6	4.0	3.724	106.05	116.66	30	118.2	1,299					16	8	0.30	24	119.08	64	30	1,491
4	2001	3	7, 8, 9				9	7.0	6.517	108.85	119.74	30	118.2	1,299					28	14	0.52	42	122.38	70	30	1,509
5	2002	3	10, 11, 12				12	10.0	9.310	111.64	122.80	31	122.1	1,342					40	20	0.74	60	125.66	76	31	1,570
6	2003	3	13, 14, 15				15	13.0	12.103	114.43	125.87	32	126.1	1,386					52	26	0.96	78	128.95	82	32	1,632
7	2004	3	16, 17, 18	120		102.33	18	16.0	14.896	117.23	128.95	33	130.0	1,429					64	32	1.19	96	132.26	88	33	1,697
8	2005	4	19-22	90	34.51	136.44	22	19.5	18.154	154.59	170.05	43	169.4	1,862					78	39	1.45	117	173.62	95	43	2,147
9	2006	4	23-26				26	23.5	21.878	158.32	174.15	44	173.4	1,905					94	47	1.74	141	178.01	103	44	2,214
10	2007	4	27-30				30	27.5	25.602	163.04	179.34	46	181.2	1,991					110	55	2.04	165	183.50	111	46	2,324
11	2008	4	31-34				34	31.5	29.327	165.77	182.35	46	181.2	1,991					126	63	2.34	189	186.81	119	46	2,348
12	2009	4	35-38				38	35.5	33.051	169.49	186.44	48	189.1	2,078					142	71	2.63	213	191.19	127	48	2,459
13	2010	4	39-42				42	39.5	36.775	173.18	190.50	48	189.1	2,078					158	79	2.93	237	195.55	135	48	2,483
14	2011	4	43-46				46	43.5	40.498	176.94	194.63	49	193.1	2,122					174	87	3.23	261	199.98	143	49	2,551
15	2012	4	47-50				50	47.5	44.222	180.66	198.73	51	200.9	2,208					190	95	3.52	285	204.37	151	51	2,661
16	2013	4	51-54				54	51.5	47.946	184.39	202.83	51	200.9	2,208					206	103	3.82	309	208.77	159	51	2,729
17	2014	4	55-58				58	55.5	51.670	188.11	206.92	53	208.8	2,252					222	111	4.12	333	213.16	167	53	2,753
18	2015	4	59-62				62	59.5	55.394	191.83	211.01	53	208.8	2,252					238	119	4.41	357	217.54	175	53	2,820
19	2016	4	63-66				66	63.5	59.118	195.56	215.12	55	216.7	2,381					254	127	4.71	381	221.95	183	55	2,930
20	2017	4	67-70	90		136.44	70	67.5	62.842	199.28	219.21	55	216.7	2,381					270	135	5.01	405	226.67	191	55	2,954
21	2018	5	71-75	72		170.55	75	72.0	67.032	237.58	261.34	66	260.0	2,857	5	70	2.65	210	288	144	5.34	432	269.33	214	66	3,499
22	2019	5	76-80				80	77.0	71.687	242.24	266.46	68	267.9	2,944					308	154	5.71	462	274.82	224	68	3,616
23	2020	5	81-85				85	82.0	76.342	246.89	271.58	69	271.9	2,988					328	164	6.08	492	280.31	234	69	3,690
24	2021	5	86-90				90	87.0	80.997	251.55	276.71	70	275.8	3,031					348	174	6.45	522	285.81	244	70	3,743
25	2022	5	91-95				95	92.0	85.652	256.20	281.82	72	283.7	3,118					368	184	6.83	552	291.30	254	72	3,880
26	2023	5	96-100	72			100	97.0	90.307	260.86	286.95	73	287.6	3,160					388	194	7.20	582	296.80	264	73	3,952
27	2024	5	101-105				105	102.0	94.962	265.51	292.06	74	291.6	3,204					408	204	7.57	606	302.20	274	74	4,020
28	2025	5	106-110				110	107.0	99.617	270.17	297.19	75	295.5	3,247					428	214	7.94	642	307.78	284	75	4,099
29	2026	5	111-115				115	112.0	104.272	274.82	302.30	77	303.4	3,334					448	224	8.31	672	313.26	294	77	4,216
30	2027	5	116-120				120	117.0	108.927	279.48	307.43	78	307.3	3,377					468	234	8.68	702	318.76	304	78	4,289
						4093.25		1562.5	1455	5548	6103	1549	6103	67,067	128	1792	67.84	5376	6250	3125	115.88	9369	6291.49	4917	1549	81,812

NOTES:

(1) SATELLITE MASS, 26.01 x 10^6 + 30% GROWTH = 33.82
OPER. MAINT. BASE, 0.122 x 10^6 + 30% GROWTH = 0.16
SCB SPARES, 0.100 x 10^6 + 30% GROWTH = 0.13
TOTAL COTV DELIVERED MASS FOR CONSTRUCTION OF EA. SATELLITE 34.11 x 10^6 KG

(2) SATELLITE MAINT. MASS (SPARES), 0.716 + 30% GROWTH = 0.931 x 10^6 KG/SAT.-YEAR

(3) COTV P/L CAPACITY = 3.94 x 10^6 KG, HLLV P/L CAPACITY = 0.091 x 10^6 KG; 3 HLLV FLTS REQ'D FOR POTV (CREW MODULE + 2 STAGES)

(4) 49/48 x 0.037095 x 10^6 KG = 0.03787 x 10^6 KG UP-PL PER 49 CREWMEN; 1 POTV FLT REQ'D FOR 48-49 CREWMEN ROUND TRIP (LEO-GEO-LEO), 14 FLTS REQ'D FOR 68-MAN CONSTRUCTION CREW

(5) 48 CREWMEN @ 0.037095 x 10^6 KG PER FLIGHT

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Subsequent to constructing and placing the LEO and GEO facilities into operation, the satellite construction program is implemented. Construction material for each satellite is transported to LEO in 409 separate HLLV flights described in Section 9.7. COTV's, previously used for GEO base build-up, will be available in LEO for direct loading (without using a depot for interim holding) of the first satellite construction mass. For subsequent satellites, the HLLV flights will be scheduled to coincide with COTV availability to permit direct cargo transfer and a COTV departure for GEO approximately every five days.

When loaded, each COTV commences the transit to GEO. The LEO-GEO-LEO cycle requires 162 days, (133 to GEO, 5 cargo transfer, 5 to LEO, 5 unloading, 9 refurbishment, 5 reloading). Upon reaching GEO, a small interorbital transfer vehicle (IOTV), Figure 9.2-2, is used to transfer the cargo directly to the satellite construction facility (similar IOTV's also are used at LEO for cargo transfer). Upon COTV return to LEO, argon tanks and thruster grids are replaced, and any unscheduled maintenance accomplished prior to rescheduling.

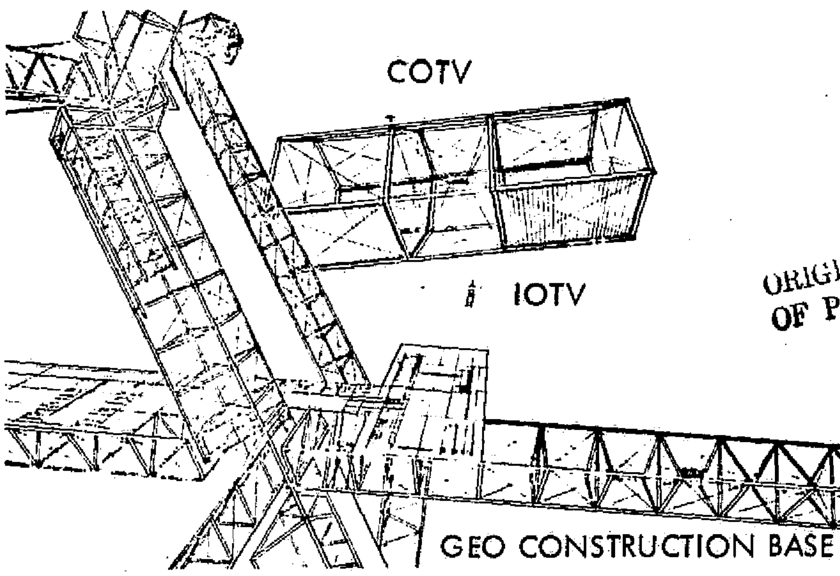

Figure 9.2-2. GEO Cargo Transfer Operations

Satellite construction and maintenance crews are carried to LEO by HLLV's, utilizing crew modules which can accommodate 48 people each plus consumables for 90 days. The module, when mated with its propulsion stages, becomes a POTV as shown in Figure 9.2-3. The logistics profile is summarized in Table 9.2-2. The stages, two to a POTV, arrive in LEO via HLLV concurrent with crew arrival and are mated with the crew module. The chemical stages require much less transit time (about 10 hours) to GEO than do the COTV's. Therefore, their departure is subsequent to COTV departure and timed for GEO arrival at the same time as the initial COTV. Returning POTV's transport to LEO crews which have completed their 90-day GEO duty cycle. The crew module with its crew and the two spent stages are then returned to earth via HLLV's.

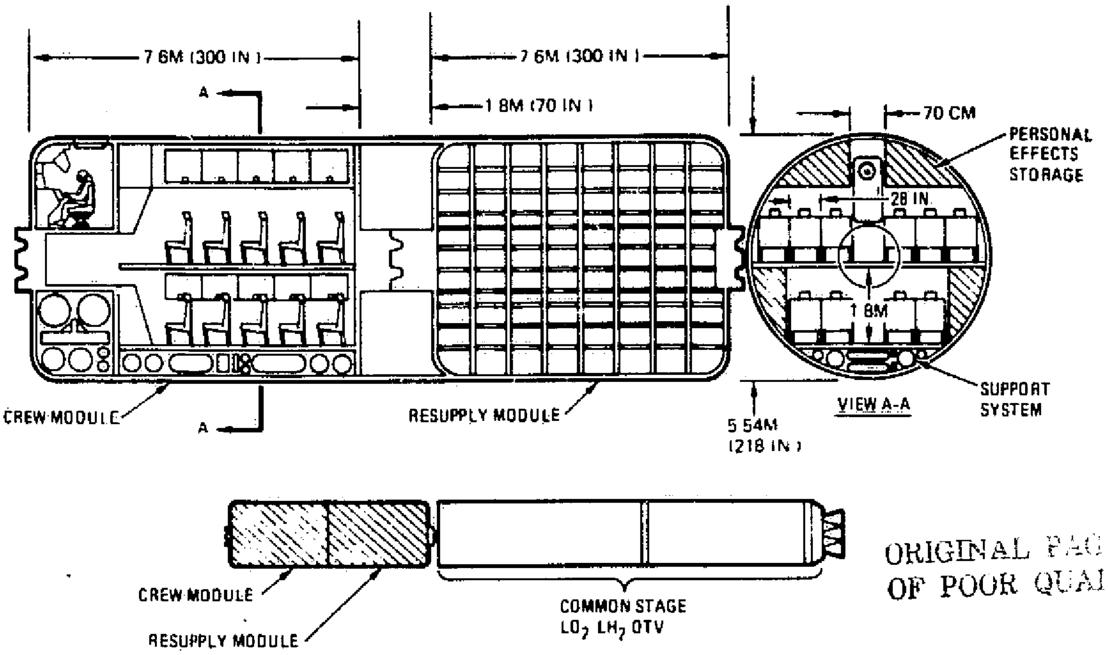

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Figure 9.2-3. Personnel Orbital Transfer Vehicle

Table 9.2-2. Crew Rotation/Resupply Logistics Profile

ITEM	FACTOR	UP PAYLOAD KG (LB)	DOWN PAYLOAD KG (LB)
PERSONNEL/PERSONAL EFFECTS	48 MEN X 110 KG/MAN	5,280	5,280
ON-ORBIT CONSUMABLES	3.6 KG/MAN-DAY X (48 MEN) X (90 DAYS)	15,550	-
CONSUMABLES CONTAINERS	10% OF CONSUMABLES	1,555	1,555
PASSENGER MODULE	200 KG/MAN X 48 MEN	9,600	9,600
RESUPPLY MODULE	20% OF CONSUMABLES	3,110	3,110
OTV CREW MODULE*	SELF-SUFFICIENT, 2-MAN CREW	2,000	2,000
TOTAL		37,095 (81,600)	21,545 (47,400)

*CONSIDERED AS INTEGRAL PART OF PASSENGER MODULES

The COTV's, after off-loading the up cargo return to LEO with packing materials, damaged equipment, and parts and consumables containers replaced as a result of maintenance operations for subsequent return to earth via HLLV. Upon arrival at LEO, the COTV's are refurbished and readied for the next transit.

Towards the completion of SPS construction, maintenance crews will be transported to the SPS via HLLV's and POTV's and will be rotated at 90-day intervals for the life of the SPS. Maintenance materials are transported to the SPS via HLLV/COTV.

Personnel involved in COTV construction or refurbishment are carried to LEO in crew modules via the HLLV.

For more detailed descriptions of the HLLV, COTV, and POTV, see Sections 4.2, 5.3, and 6.0.

9.3 LEO SUPPORT OPERATIONS

LEO operations include COTV construction and maintenance, payload transfer from HLLV to COTV and vice versa, POTV stage mating, crew transfer, and base maintenance.

Crew size required for COTV construction is shown in Table 9.3-1 and are estimated for construction of one COTV. Figure 9.3-1 contains the COTV construction sequence and timeline for one COTV. Assuming that construction of the second vehicle commences immediately upon completion of the structures, solar blanket and PDS for the first vehicle, it will take approximately 90-days to complete a set of 10 COTV's. The construction crew shift size to support this schedule would peak at 42 people. Total crew size; which includes all construction and support personnel for a 4 shifts operation totals 213.

Table 9.3-1. OTV Construction Crew Functions and Size

	Shift Size	Crew Size (1)(2)	
Construction Crew	(36)	(144)	
Primary Structure	18	72	
PDS, Solar Blankets, & Att. Control	18	72	
Construction Support	(4)	(16)	
Construction Pwr Management	1	4	
Logistics & Inventory Control	1	4	
Intrafacility Vehicles	2	8	
Management	(4)	(16)	
Administration	1	4	
Security & Safety	1	4	
Communications	2	8	
Crew Support	(3)	(13)	
Food Management	3	12	
Medical/Recreation	–	1	Total
	47	189	

(1) Based on 8 hour/shift, 3 shifts/day, 8 work day cycle (6 on, 2 off) and 4 shift crews

(2) Estimated crew size for constructing 1 EOTV vehicle

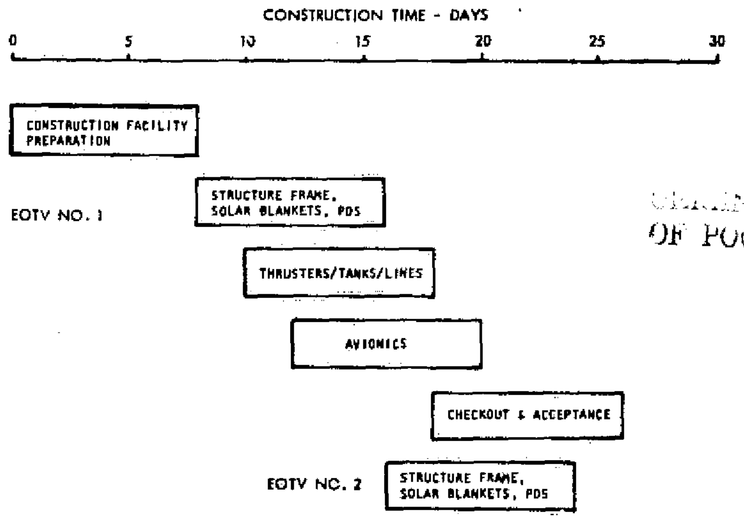


Figure 9.3-1. COTV Construction Timeline

LEO support operations will require a permanent crew of 30 at the LEO facility shown in Figure 9.3-2. The LEO base personnel provide supervisory activities for transfer of up and down payloads between the HLLV and the OTVs, and perform the scheduled maintenance required by the COTV (changeout of thruster screens and argon propellant tanks). It has one crew hab and one crew support module of the same configurations as the GEO construction base. Direct transfer of crew and equipment between the HLLV and the OTV's are planned; however, multiple docking ports and excess subsystems capacity and power are provided for emergency staging support. This crew, in addition to maintaining COTVs, will support payload transfer operations for GEO payloads and mating of crew modules and POTV stages. This operation is on-going. Activities pursuant to COTV construction at the LEO facility are more sporatic in nature but will require additional facilities for the larger construction crew.

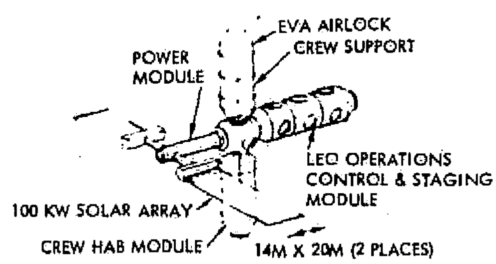


Figure 9.3-2. LEO Base Concept

The first COTV set will be required in time to support construction of the GEO facility; later it will be used to support satellite construction through the third year of the program. Three satellites are scheduled for construction during the fourth year and will require the support of two COTV sets. Therefore an additional set must be constructed in time to commence LEO-GEO transit during the twelfth month of the third year.

Two COTV sets can provide the required support for the mature program construction rate of 4 satellites per year. An additional set will be required to support the ultimate construction rate of 5 per year.

At the end of each operational year, 1.02×10^6 kg of spares must be provided to each satellite. During the first three years, these spares can be carried in the COTV sets used for transporting satellite construction material, since the 10th COTV is not fully loaded. However, an extra EOTV (1 vehicle) must be available at the beginning of the fourth program year to transport the increasing mass of satellite maintenance material to GEO. Each COTV can make 2+ round trips per year carrying about 3.9×10^6 kg of mass per trip. At the beginning of the sixth program year, when there are 11.5 satellites requiring maintenance, one more COTV must be provided. Towards the end of the 30 years, the annual maintenance requirements will approach 122×10^6 kg, requiring approximately 1½ COTV sets (15 COTV's) above the basic 3 sets supporting construction. This results in an average construction rate of one vehicle every two years.

Since the LEO base facilities required to support the permanent 30-man LEO maintenance crew are insufficient for the 1890man COTV construction crew, eight additional crew support modules will be required during the COTV construction period. After an COTV construction cycle, these modules can be detached and transported to GEO as part of a satellite maintenance base, and then replaced with new modules from earth when the next construction is scheduled. An alternative is to leave them in position and accept a facility that is oversized for normal operations. During the follow-on study, these alternatives will be evaluated in light of a more detailed COTV construction schedule and the cost differences involved to arrive at an optimum procedure.

9.4 Nth SATELLITE CONSTRUCTION SCENARIO

Identification of the major construction operations and their time-phased relationship with each other and with the overall construction schedule for a single satellite are given in Figure 9.4-1.

A single integrated construction facility builds the structure, installs the solar blankets, the reflectors, the power distribution system and other subsystem elements located in the wings. Since the mature program specifies a construction rate of one satellite every 90 days, this schedule provides sufficient time to support the program with a single construction facility. Construction starts with one wing tip and progresses toward the center section where the rotating joint for the MW antenna is to be located, and thence continues outbound building wing No. 2 and terminating at that wing tip.

The first eight days are designated for preparation of the construction facility. Prior to the eight day sufficient materials have been delivered by the COTV to satisfy the first several days of construction: primary structure material (beam machine cassettes) for 1/2 the satellite; solar blanket and

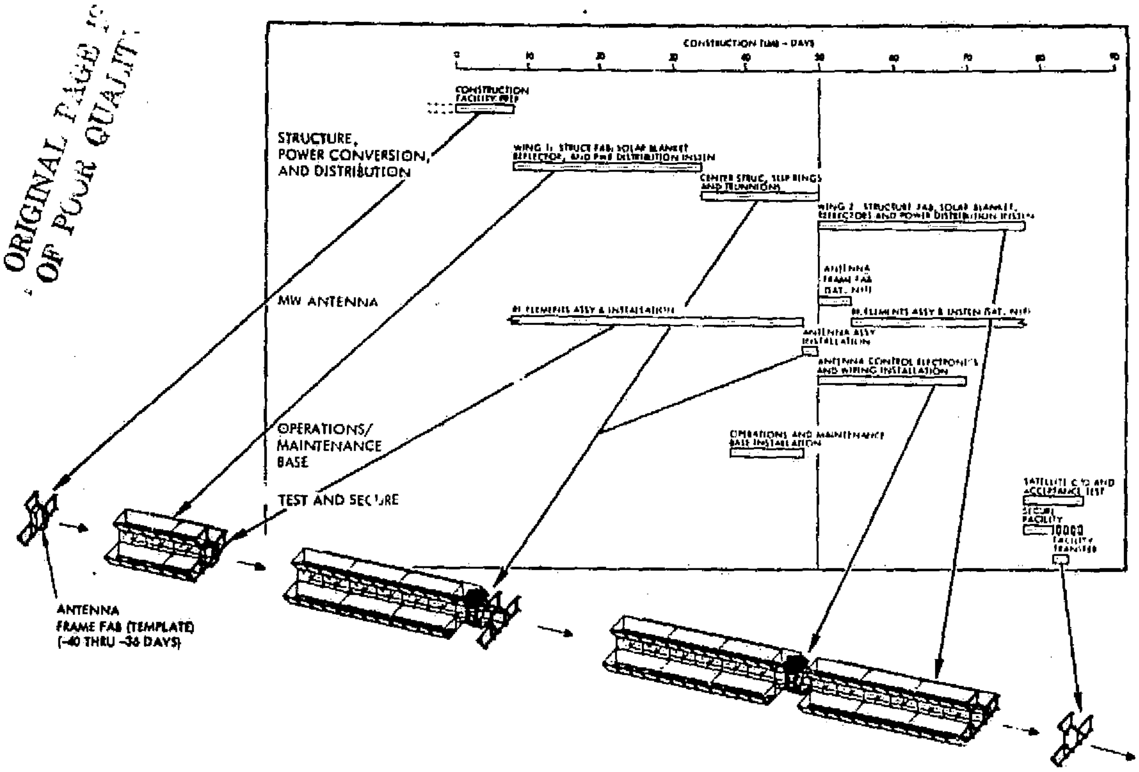
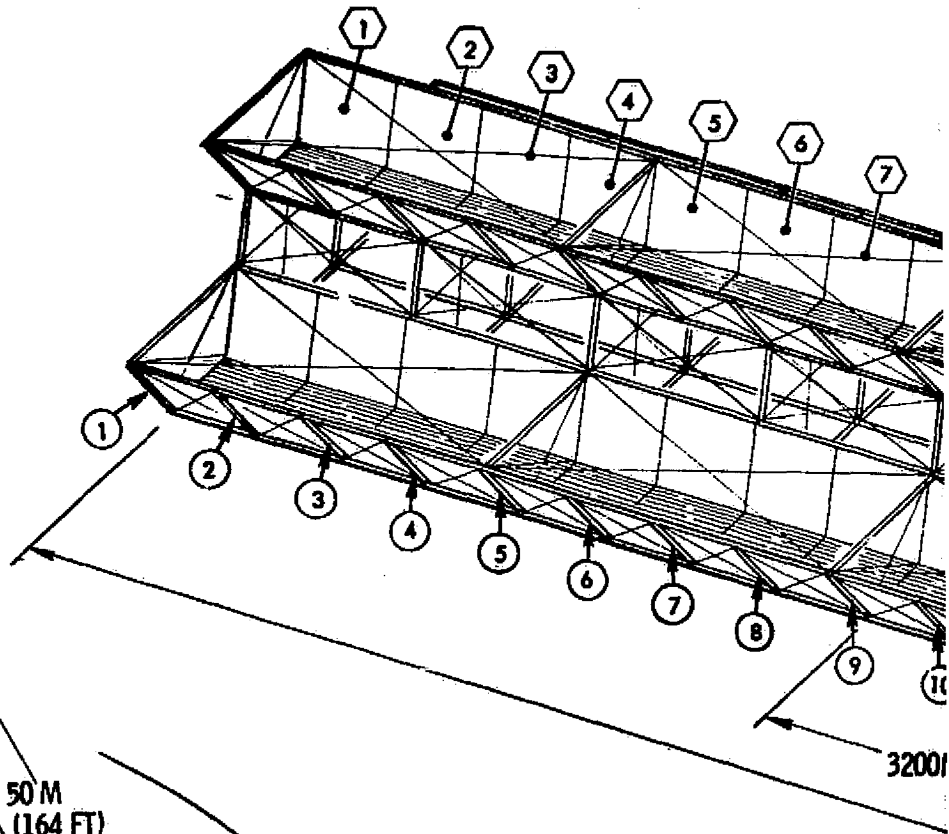
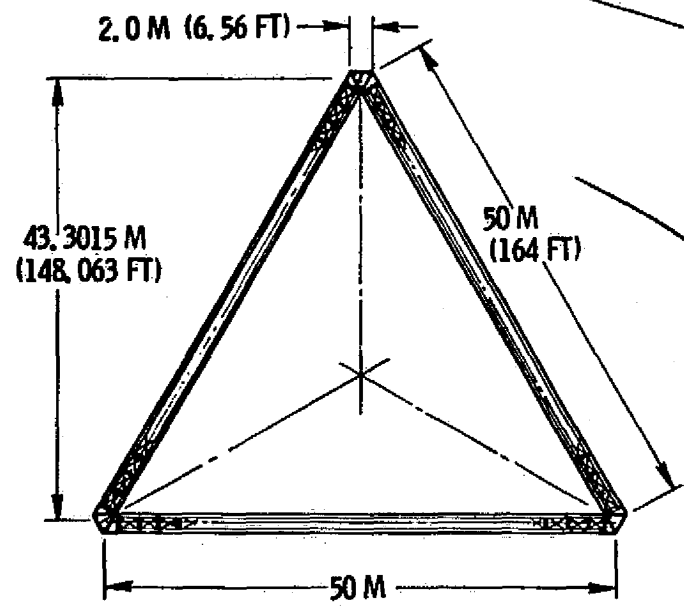


Figure 9.4-1. Nth Satellite Construction Sequence

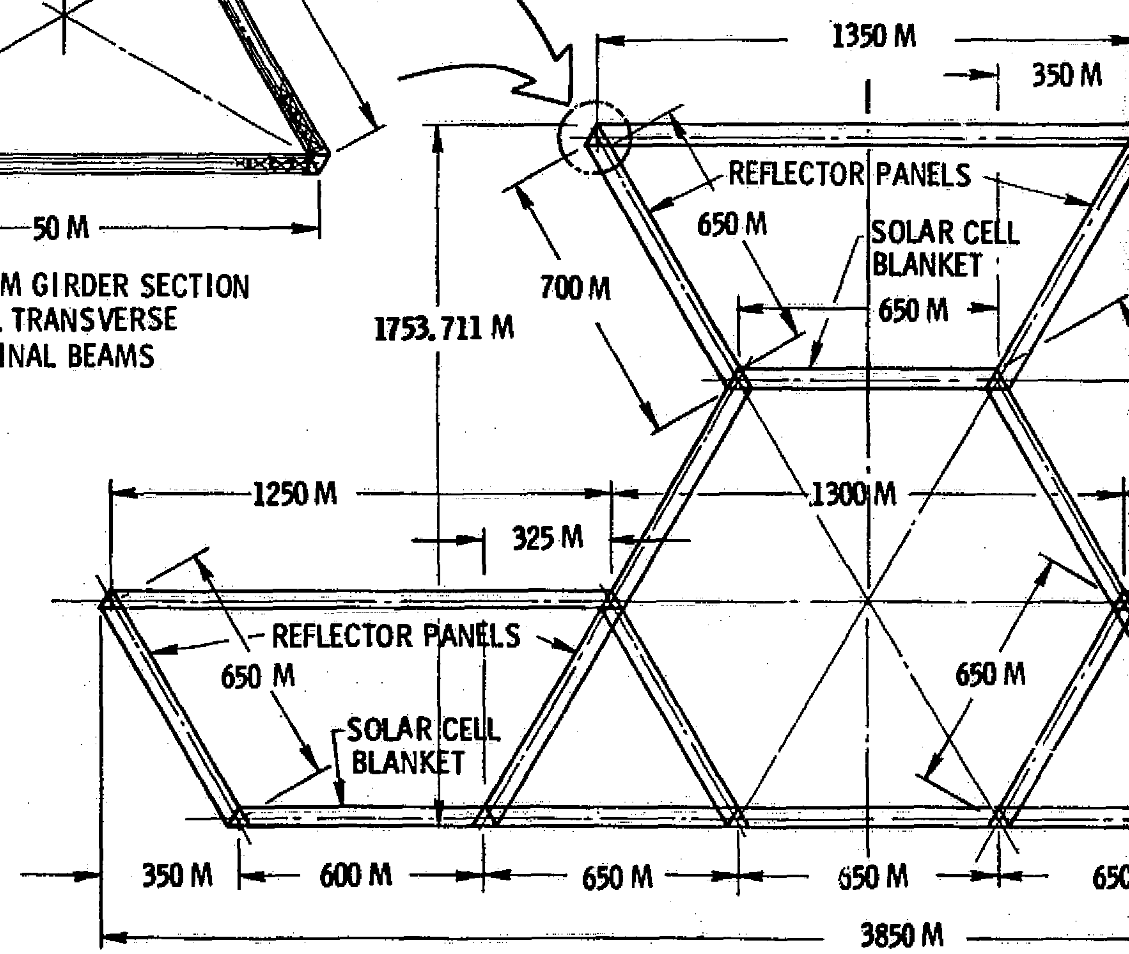
reflector rolls, electrical conductors and switch gear for the first two bays; and MW antenna components. Since the rear side of the facility is always exposed to space with no interference from the main construction activities, it is implemented as the jig for building the MW antenna frame and as the location for assembly, and installation of the 30m x 30m rf mechanical modules. Fabrication of the MW antenna for this Nth satellite was started on the 50th day of construction of the previous (N-1) satellite and is continued up through the 48th day of construction of this satellite; at that time it is ready for installation into the slip-ring-mounted trunions.

Each satellite wing consists of 12 bays 800 m long numbered as shown in Figure 9.4-2. These are constructed at the rate of one every two days using three 8-hour shifts of 78 men per shift. Table 9.4-1 shows shift utilization for a two day period. Prior to the start of longeron fabrication, the solar array blankets and reflectors for one bay are placed in position for deployment and attached to the frame of the preceding bay so that they may be unrolled as beam fabrication progresses. Similarly, PDS switch gears are installed on the frame and main feeders positioned for unrolling. These operations, requiring 39 men per shift, are accomplished during the first two and one-half shifts. During beam machine operation the same crew installs and fastens the various rolls to longerons and cables as they deploy.





50 M TRIBEAM GIRDER SECTION
TYPICAL ALL TRANSVERSE
& LONGITUDINAL BEAMS



SATELLITE POINT DESIGN STRUCTURAL CROSS

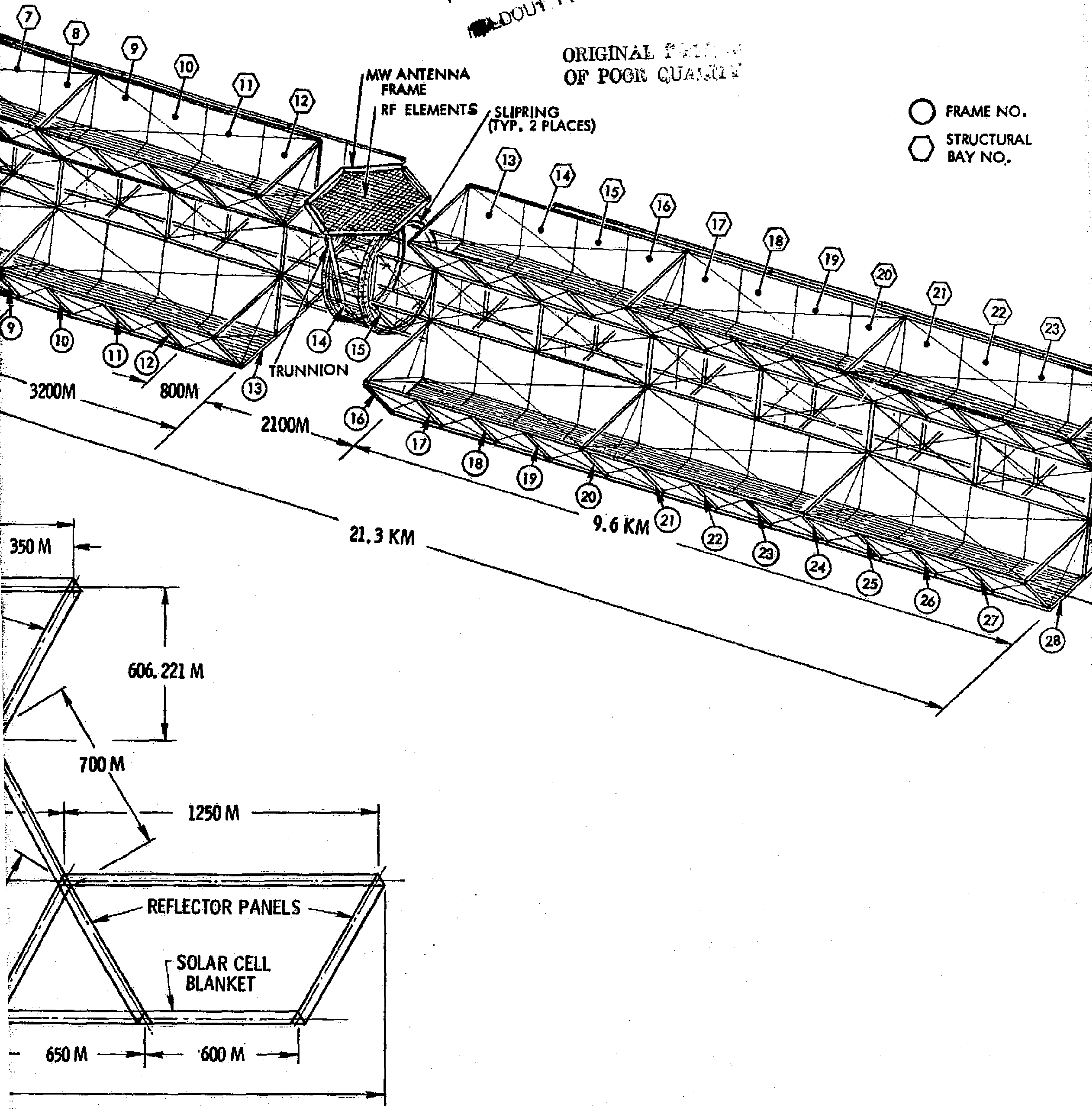


Figure 9.4-2. Point Design Satellite Configu

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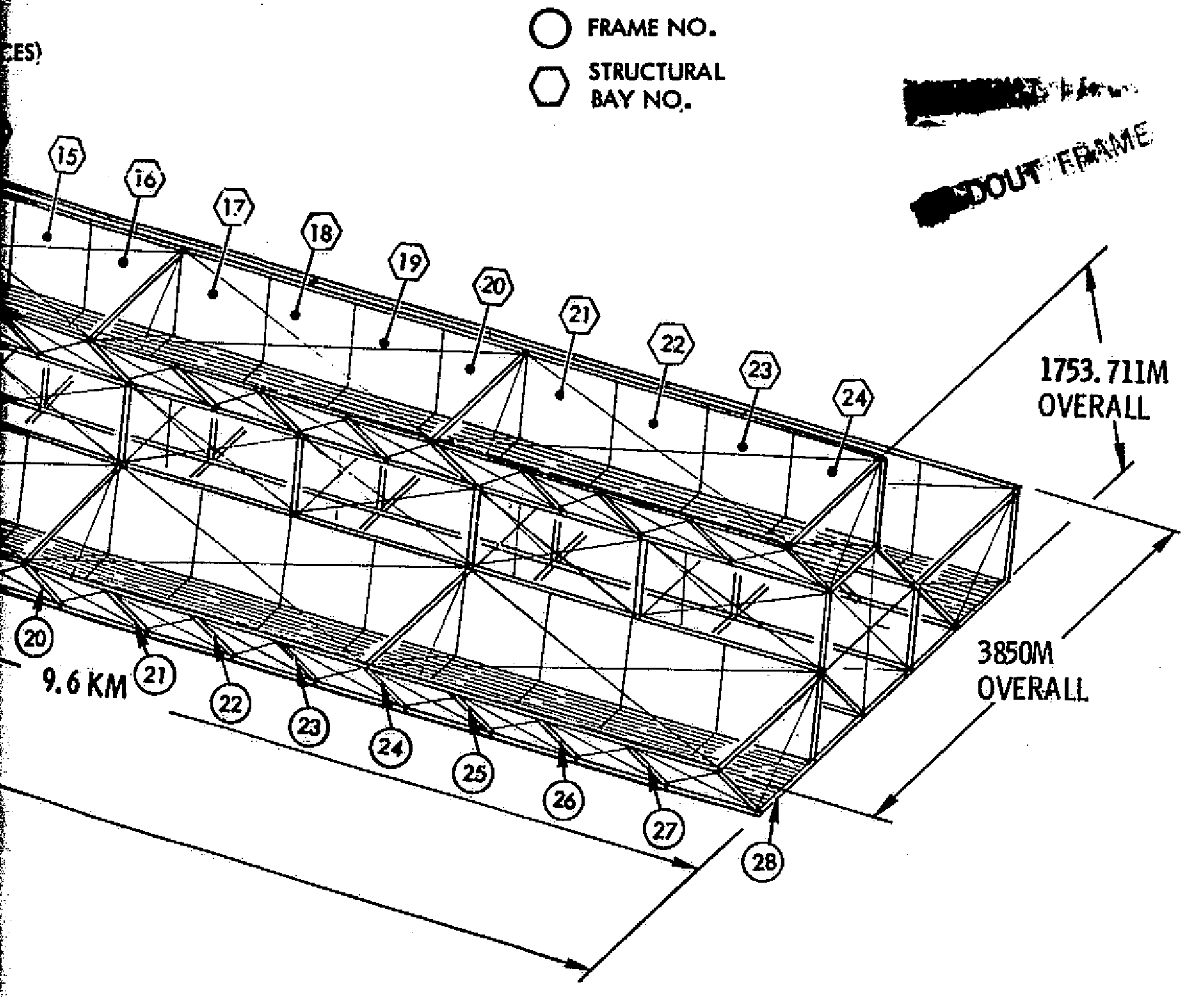

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Figure 9.4-2. Point Design Satellite Configuration

Table 9.4-1. Shift Utilization
– Bay 3, Frame 4

	SHIFT 1		2		3		4		5		6	
STRUCTURE												
FRAME, 3 COMPLETE												
CONSTRUCT LONGERONS						18	18	6	6			
CONSTRUCT TRANSVERSE BEAMS						18	18					
MFG & INSTALL END FITTINGS TO BEAMS								36	42			
INSTALL TRANSVERSE BEAMS										42	42	
ALIGNMENT											12	24
SOLAR BLANKETS												
ATTACH BAY @ END CAT. TO FRAME 3	18											
INSTALL ROLLS AND CABLES		6	6									
ATTACH BAY 3 ROLLS TO FRAME 3			12	18	12							
UNROLL S/A AND CABLE—ATTACH TO LONGERON FOR DEPLOYMENT & CHECKOUT						12	12	12	12	12	C/O 6	12
REFLECTORS												
ATTACH BAY 2 END CAT. TO FRAME 3		18										
INSTALL AND UNFURL ROLL			24	24								
ATTACH BAY 3 CAT. TO FRAME 3 LEADING EDGE					18							
UNROLL REF. & CABLE—ATTACH TO LONGERON						12	12	12	12			
PDS												
INSTALL/CONNENCT SWITCH GEAR—BAY 2—CHECK OUT	18	18										
CONNECT SWITCHES TO FEEDERS & SOLAR ARRAY, CHECK OUT		18										
INSTALL FEEDER INSULATION MOUNTS			18	18	24							
ROLL OUT & ATTACH FEEDERS TO LONGERON, CHECK OUT					6	12	12	12	6	6	C/O 6	12
INSTRUMENTATION AND CONTROLS												
INSTALL	36	36	18	18	18	6	6					
CHECKOUT										12	12	12
SHIFT SIZE	72	78	78	78	78	78	78	78	78	78	78	60

The longerons are fabricated autonatically during 2 shifts starting in the middle of shift No. 3. The beam machines produce longerons at the rate of 2 m/minute, or 800 meters in approximately one 8 hour shift. The operation is spread over 2 shifts to allow for fastening of blankets, cables, etc., as the longeron advances. The transverse beams are fabricated during the first of the two shifts noted above. During the next shift, transverse beam machines are shutdown, end fittings are added and the beams are installed in position. This latter operation requires translation of the beam machines out of the beam plane to permit fabrication and attachment of the beam end fittings. The beams are then translated into position and fastened to the longerons. Installation of transverse beam end fittings, beam translation, and securing in place are considered to be manual EVA operations requiring 3 men at each beam end, which utilizes 42 of the total 78-man shift crew. Personnel are stationed at each of the beam machine stations during machine operation. However, since all beam machines are shutdown during transverse beam joining operations, this permits use of these crews for beam fitting and fastening.

The entire wing structure and the power conversion system (solar blankets, reflectors and power distribution system), is completed on the 34th day. (More detailed descriptions of the fabrication and assembly procedures is contained in Section 9.4.3.) While the wing No. 1 construction is taking place, the MW antenna crews are proceeding with the assembly, test, and installation of the antenna elements into the antenna frames. The antenna assembly continues during construction of the center section. (Section 9.4.3 contains a description of antenna assembly procedures.)

Subsequent to completion of wing No. 1 the construction facility constructs the longerons and frames in the center section, installs the sliprings, constructs the trunion supports, installs the turnions, and installs power wiring in the center. Although 16 days are scheduled for this activity, the timeline requires only 12 days with two additional days scheduled for transfer of the antenna to the trunion mounts. Two days are allowed for contingencies. Immediately upon completion of the center section primary structure the facilities for the operation and maintenance base are installed and the first operational maintenance crew arrives to support installation of the antenna control electronics and satellite checkout, which takes place from day 50 through day 69.

By the 51st day all satellite hardware has been delivered. On site logistics activities are therefore greatly reduced freeing construction support personnel for subsystems hookup and checkout during the wing No. 2 construction period.

Use of the construction facility is completed on day 78 and flyaway transfer to the construction site of the next satellite occurs on day 84. Final satellite checkout and acceptance testing is completed on day 86.

9.4.1 GEO CONSTRUCTION BASE CONCEPT

Construction of the satellites takes place in GEO, each satellite being constructed at its designated longitudinal location. All construction activities are supported by a single integrated construction base which produces satellites at the rate of 4 per year during the mature portion of the program. Upon completion of one satellite the base is moved to the operational location of the next satellite for construction of that satellite.

The construction base Figure 9.4-3 consists of the satellite construction fixture, the construction equipment, and the base support facilities and equipment. The construction fixture is a rugged heavy gage metal structure on which all elements of the construction base are mounted. The fixture constitutes the reference surfaces for the construction operations and the locating jig for the equipment which constructs/installs various elements of the satellite in situ.

The major construction equipment includes the 50 m tribeam fabricators; the deployment equipment for the solar cell blankets, the solar reflector panels, the power distribution conductors, the cables for retention of the solar blankets, and the structure tensioning cables; the assembly facility for the MW antenna mechanical modules; and the equipment for installation of the MW antenna elements into the antenna frame. The location of most of these elements are identified on Figure 9.4-3.

GEO construction base support facilities and their locations also are identified in Figure 9.4-3. A crew size requirement of 680 has been estimated for accomplishing the construction in the scheduled time. The crew and their facilities are divided equally and are located on each side of the hex portion of the fixture as shown. One of these 340 men facilities shown in more detail in Figure 9.4-4, consists of 7 three-module crew habitability complexes plus 2 base management modules, 2 pressurized storage modules and solar array power modules.

The modules of the crew habitability complex are described in more detail in the lower right of the figure. Each complex is composed of two of the crew hab modules, each of which provide staterooms, personal hygiene facilities and support subsystems for 24 crew members; and one crew support module which provides galley, recreational and medical facilities and subsystems for the 48 crew members of the two crew hab modules. The base management modules house the communications and control systems for the construction base. The pressurized storage modules include workshops for maintenance of construction facility elements and satellite hardware as required.

Seven of the modules (enclosed by the dashed lines) are hardened against solar flare radiation and serve as temporary quarters for the entire crew when the base is subjected to that environment.

(1) CONSTRUCTION FIXTURE
(2) BASE SUPPORT FACILITIES & EQUIP (2 EA)
340-CREW MEMBER SUPPORT
BASE SUBSYSTEM, MAINTENANCE SHOPS
BASE MGMT, COMM., CONTROL, LOGISTICS
(3) WAREHOUSE
(4) EOTV DOCKING/CARGO RECEIVING
(5) POTV DOCKING
(6) MW ANTENNA ASSEMBLY FACILITIES
(6A) FRAME FABRICATION FIXTURE
(6B) RF ELEMENTS ASSY & INSTL FACILITY
(6C) FRAME TRANSLATION GUIDEWAY
(6D) FRAME (50-M TRIBEAM) FABRICATORS
(7) SATELLITE FRAME (50-M TRIBEAM) FABRICATORS (33 PLCS)
(7A) LONGERONS (14 PLACES)
(7B) TRANSVERSE FRAME BEAM (19 PLACES)
(8) BEAM FAB/INSTALLATION WORK STATIONS (14 PLCS)
(9) SOLAR BLANKET & PDS INSTL STA.
(10) SOLAR REFLECTOR PREP/INSTL STA.
(11) INTRA-BASE LOGISTICS VEHICLES

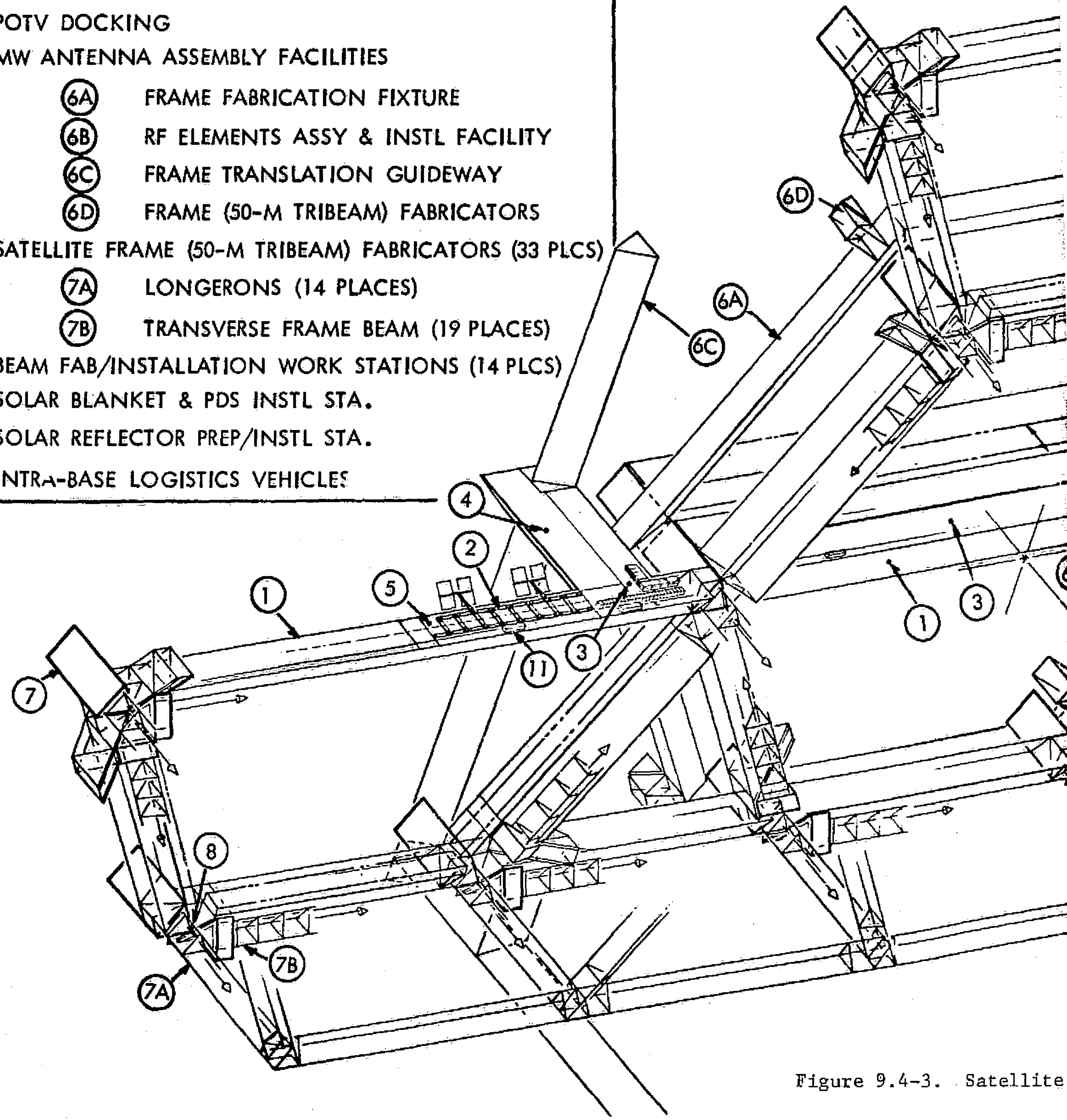

Figure 9.4-3. Satellite

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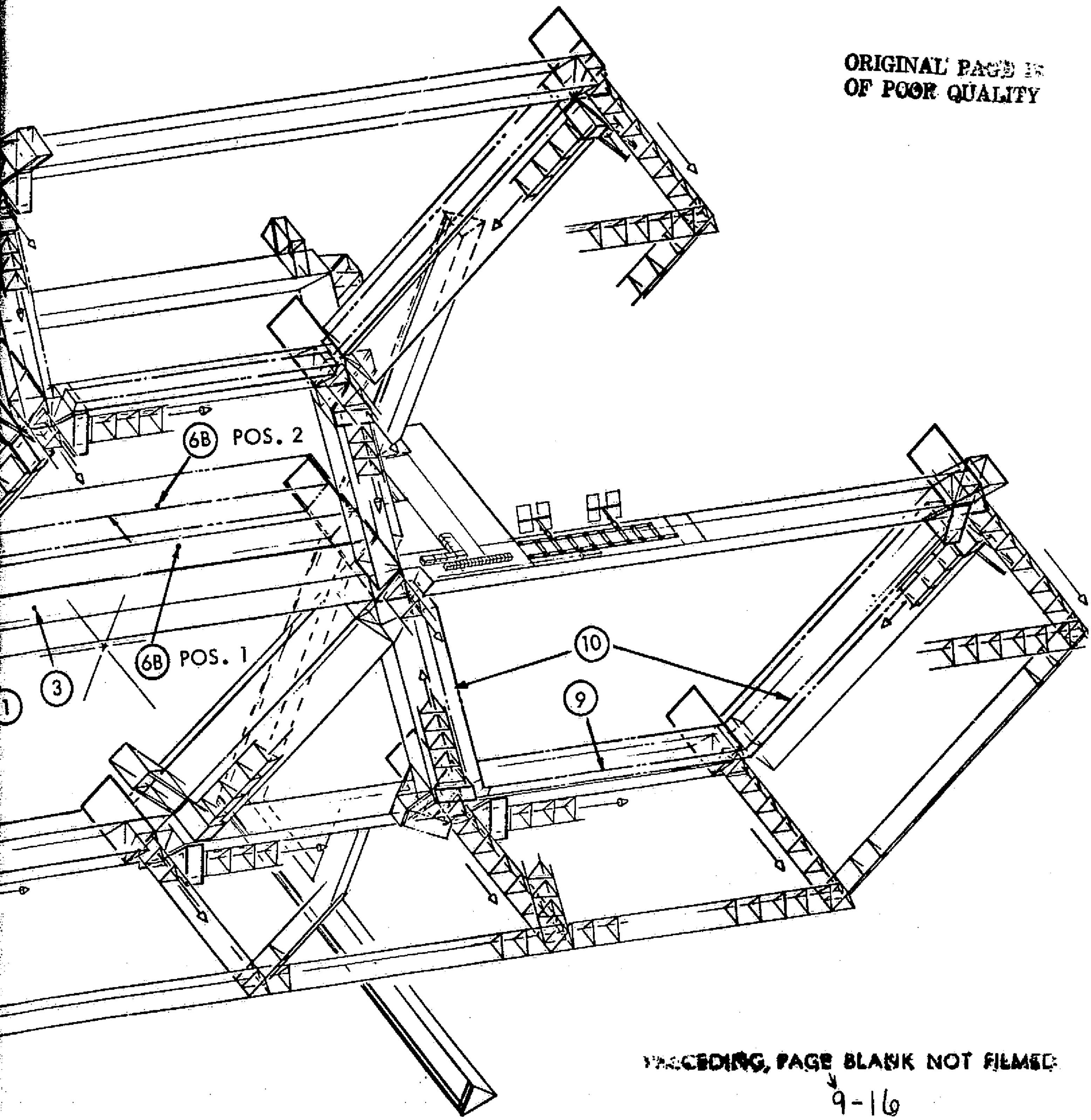

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. Satellite Construction Base (SCB)

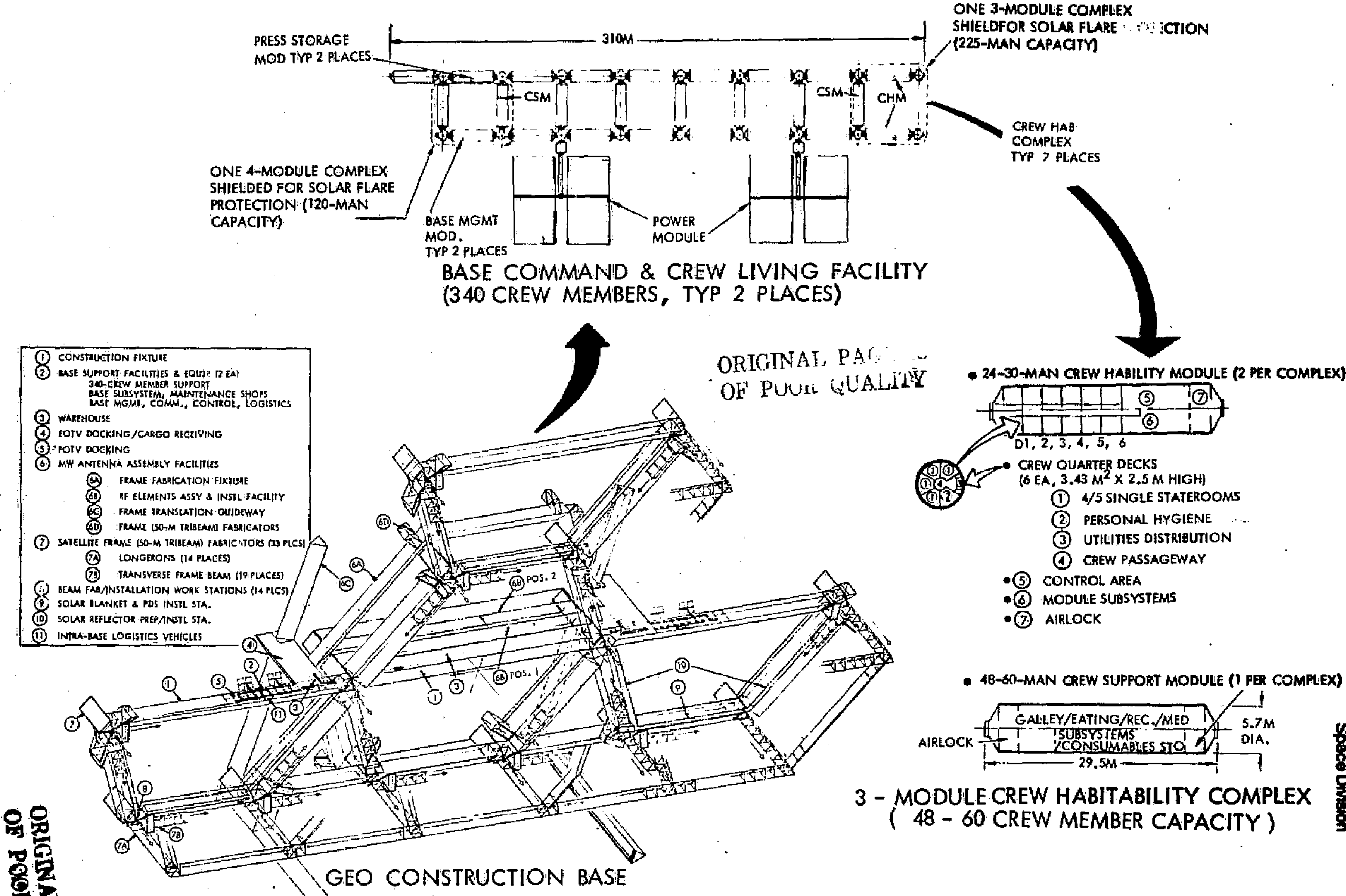

Figure 9.4-4. GEO Construction Base Support Facility

9.4.2 SATELLITE CONSTRUCTION CREW SIZE

Personnel requirements for GEO construction operations include both the manpower needed to fabricate and checkout the satellite, and the facility crew which provides the necessary construction support functions.

The satellite construction sequence has been described in Section 9.4 and Figure 9.4-1. A crew shift size of 78 is required to construct the basic wing structure and install and fasten the solar array blankets, reflectors, and PDS. Crew utilization for this operation is more fully discussed in Section 9.4. Upon completion of the first wing, this crew then fabricates and installs the center structure, slip-rings, and trunnions before starting construction of the second wing. Construction of the microwave antenna frame, installation of waveguide panels and klystrons, and installation of the completed antenna is accomplished by the 50th day by a crew of 32 per shift. After the antenna is installed, this same crew commences work on the antenna for the next satellite. Installation of antenna control electronics is done by the satellite maintenance crew of 20, which arrives on station at approximately the 50th day. Upon completion of the satellite, the construction crew conducts checkout operation and readies the construction facility for the next satellite. GEO base support functions include construction support (power management, logistics, and intra-facility vehicles), maintenance of construction equipment, vehicles, and base subsystems, base management, and crew support. The shift size for accomplishing these functions is 60 people.

Total crew size for the overall operations is 680. This is based on three 8 hour shifts per day and an 8 day work cycle (6 days on, 2 days off) resulting in a 4 shift crew. Table 9.4-2 shows shift size and crew size, broken down by both construction and base operations functions. These figures will be updated as required to reflect any changes in the point design affecting the construction procedures.

9.4.3 ASSEMBLY EQUIPMENT AND OPERATIONS

Primary Structure Fabrication

The overall satellite structural configuration, shown in Figure 9.4-2, is comprised of two 9600-m long wings, each divided into twelve 800-m bays separated by 13 transverse frames. Referring to the cross section in the lower left of the figure, the structure provides for three troughs: top center, lower left and lower right. 650-m long cross beams form the bottom of the top trough and 600-m long cross beams form the bottom of the two side troughs. The solar blankets are located on the bottom of the troughs, while the reflectors are on the diagonal sides. Longerons are located at the intersection of the various cross beams and diagonals; the six longerons forming the central hex section are continuous for the entire length of the satellite; the remaining eight longerons are continuous for the length of each wing.

The satellite primary structure is constructed of 50 m tribeam girders which utilize the basic 2 m triangular beam elements for the longitudinal elements at the three corners and for the transverse ties which occur at 50 m

Table 9.4-2. Satellite Construction Crew Function and Size

	SHIFT SIZE	CREW SIZE (1)
CONSTRUCTION CREW	(110)	(440)
Primary Structure	39	156
PDS, Solar Blankets	39	156
Reflectors & Attitude Control	32	128
CONSTRUCTION SUPPORT	(18)	(72)
Construction Power Management	6	24
Logistics and Inventory Control	9	36
Intra Facility Vehicle	3	12
MAINTENANCE	(11)	(44)
Construction Equipment	5	20
Vehicles	3	12
Construction Base Systems	3	12
BASE MANAGEMENT	(13)	(52)
Administration	7	28
Security and Safety	3	12
Communications	3	12
CREW SUPPORT	(18)	(72)
Food Management	12	48
Medical/Recreation	3	12
Housekeeping	3	12
TOTALS	170	680

(1) Based on 8 hours/shift, 3 shifts/day, 8 work day cycle (6 on, 4 off) and 4 shift crews.

intervals, as indicated in Figure 9.4-5. The basic structural 2 m triangular beam is constructed as a continuous element by a single beam machine, Figure 9.4-6, from three prepunched ribbons of 0.254 mm thick aluminum sheet stock which are fed into the machine from cassettes as indicated. (Shear stabilization of both the tribeams and the satellite wing is achieved by use of the X-tension cables.) The 50 m tribeams are also fabricated as continuous elements. The tribeam fabricator, Figure 9.4-7, is one concept for a unitized 50 m girder builder. This concept utilizes six beam machines to construct each girder; one for each of the three longitudinal members and one for each of the three sides to fabricate the transverse ties. The six beam machines are located to provide proper spacing and alignment of the 2 m longitudinal and transverse elements; it also performs the joining operation between those elements, while producing the girders without interruption.

[A modified version of this fabricator incorporates only four beam machines; one each for the three longitudinal members and a single machine producing the cross-ties for all three sides. The machine producing the cross-ties would build its 2 m beam at the rate of 6 m/minute, or somewhat faster, to accommodate the increased capacity without slowing the overall build rate. This technique is probably more suitable to an interrupted production of 50 m tribeam girders: i. e., interruption of production of the three longitudinal 2 m members every 50 meters to install the cross-ties. Interrupted production would seem not practicable for the satellite longerons because of the acceleration forces resulting from such frequent stop-start cycles. However, it would seem appropriate for the transverse 50 m girders since their maximum length is only 1350 m (total mass of about 4600 kg)].

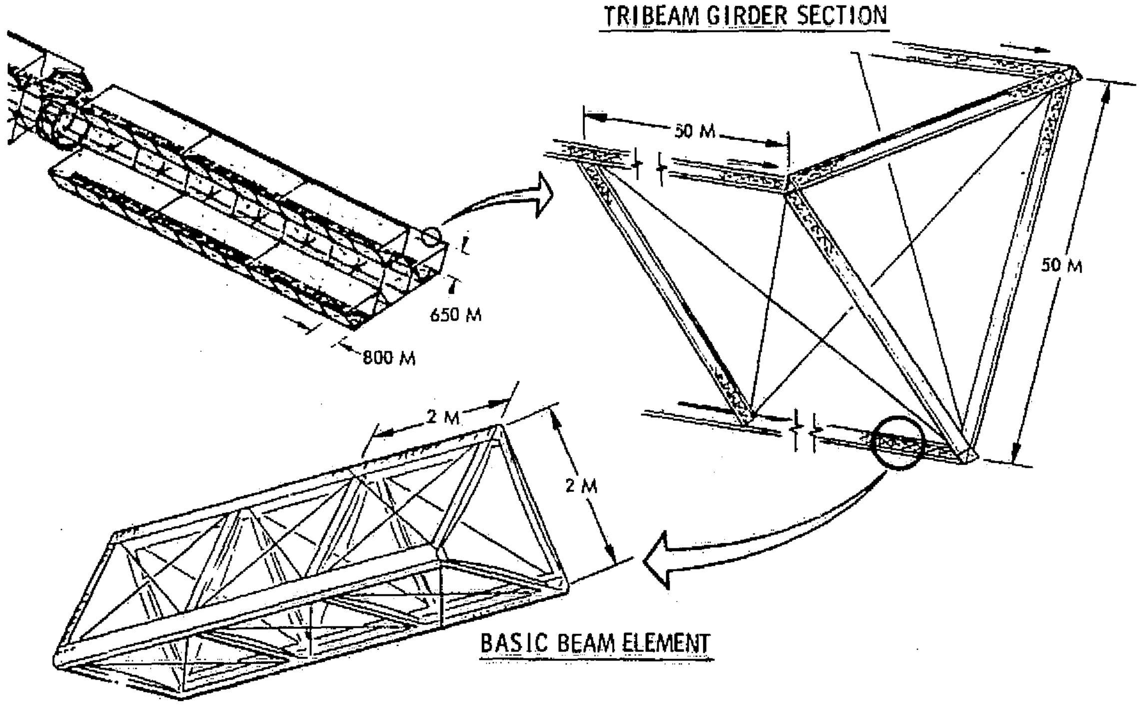

Figure 9.4-5. Primary Structure

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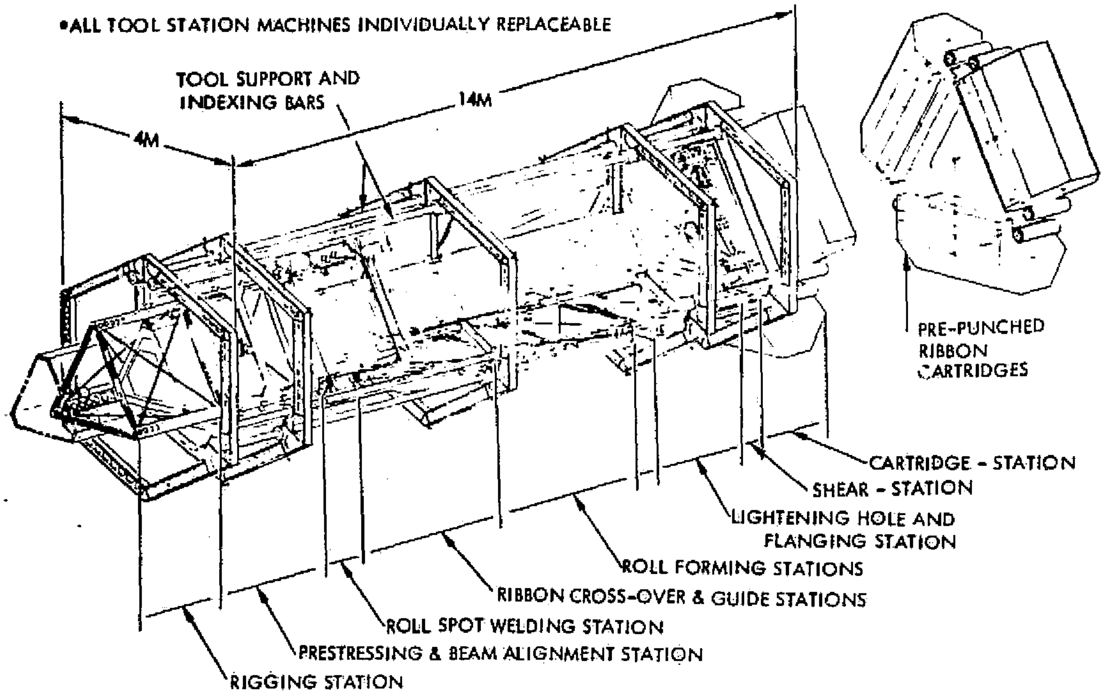

Figure 9.4-6. SPS Structural Element Fabricator

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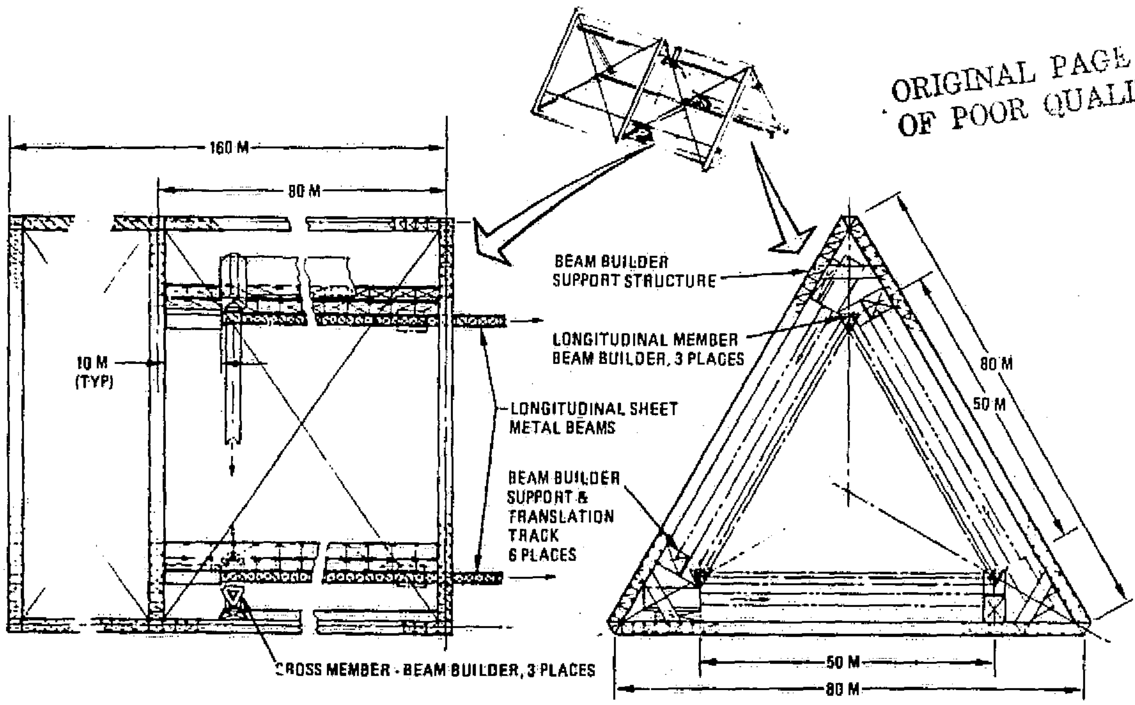
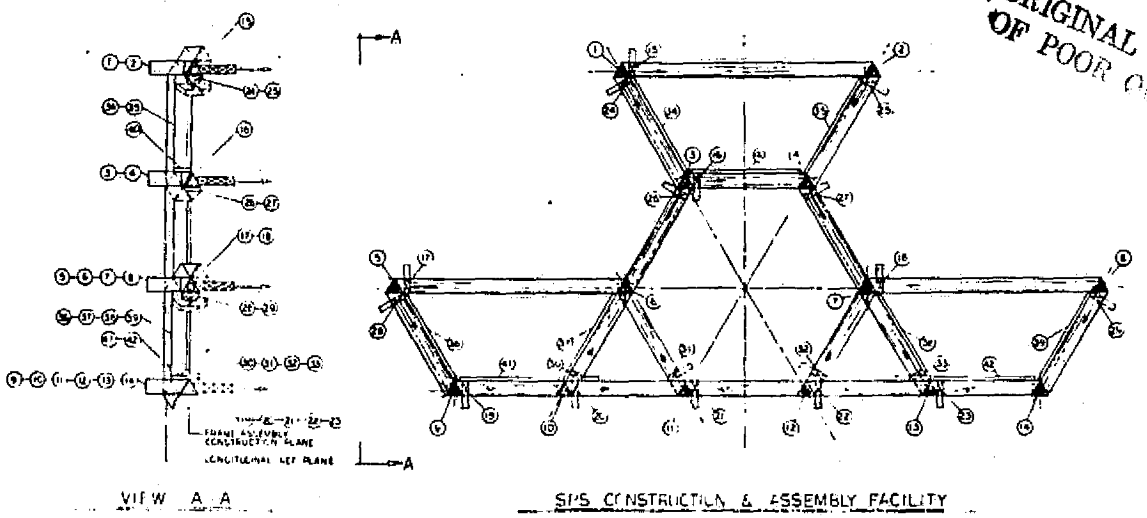


Figure 9.4-7. 50 Meter Tribeam Girder Fabricator

There are 33 tribeam fabricators used to build the primary satellite structure. They are mounted on the satellite construction fixture in the locations shown in Figure 9.4-8. Numbers 1 through 14 construct the 14 longerons and Numbers 15 through 33 construct the 19 frame segments. (Additionally there are six fabricators mounted on the back of the fabrication facility to produce the hex frames for the MW antenna.) The construction base drawing, Figure 9.4-3 provides a better perspective of the tribeam fabrication installation on the satellite construction facility and shows the direction of movement of each tribeam as it is fabricated. The construction facility is configured to restrain the free-end of each cross-frame member as it is fabricated. After completion of each 800-meters of longitudinal members, all beam machines are stopped, the cross-frame fabricators are translated to their offset positions and the cross-frame members are completed and joined to the longerons.



NUMBERS 1 THROUGH 33 REFER TO PHYSICAL LOCATIONS OF TRI-BEAM COMPLEXES
NUMBERS 34 THROUGH 39 REFER TO REFLECTOR PANEL INSTALLATION FACILITIES LOCATIONS
NUMBERS 40, 41 AND 42 REFER TO SOLAR BLANKET INSTALLATION FACILITIES LOCATIONS

Figure 9.4-8. Satellite Construction & Assembly Facility

Figure 9.4-7 indicates the fabricators are of two lengths; 80 m for the transverse frame beams and 160 m for the longerons. The added length for the longerons provides for 80 m of machine travel in the event of a hangup. This is necessary since all 14 longerons are fabricated simultaneously (at the rate of 2 m per minute) and are connected together by the frames which have already been built. If one of the longeron fabricators hangs up this allows 40 minutes to remedy the problem or to shutdown the machines before the malfunction impacts the rest of the structure. Over-travel is not necessary in the case of the transverse beams since each transverse beam fabricator operates independently and all machines (longitudinal and transverse) are shutoff at the time the transverse beams are attached to the longerons.

Solar Cell Blanket Installation

The installed configuration of the solar blankets is illustrated in Figure 9.4-9. Blanket location is indicated in Figure 9.4-2. Both the longerons and the cross beam to which the solar blankets are attached are installed "points up" (that is, with one apex of the 50 m tribeam girders up rather than with a 50 m side up). This provides maximum free area between the frames for the solar cell blankets. The blanket in each 800 m-long bay is a structurally independent installation suspended by catenaries attached to the longerons on the sides and to the cross beams on the ends. Each two bays of solar blankets are electrically connected in series, as shown on the schematic, Figure 9.4-10, thus constituting a functional module which produces the required voltage. The two-bay modules results in the requirement for 21 switch gears to be mounted on the cross beams which occur between the two bays and 11 switch gears on the (alternate) cross beams which occur between modules. Additionally secondary feeders are installed on the end-of-module cross beams to bring the current into the main feeders. The main feeder cross sectional area is increased at the output of each two-bay module to accommodate the step increases in current.

The solar blankets are installed in longitudinal strips approximately 25 m wide. In the upper trough the solar blanket is 24 strips wide and in the lower troughs 22 strips wide. Each two strips constitute a structural unit suspended by 50 m wide catenaries at the leading and trailing edges, the two ends of the leading and trailing edge catenaries attaching to the 50 m cross beams at the location of the beam cross ties which occur at 50 m intervals. The outside longitudinal edges of the two outside solar blanket strips attach to the longerons via a series of longitudinal catenaries, the ends of which pickup the cross-braces points of the longerons which occur at

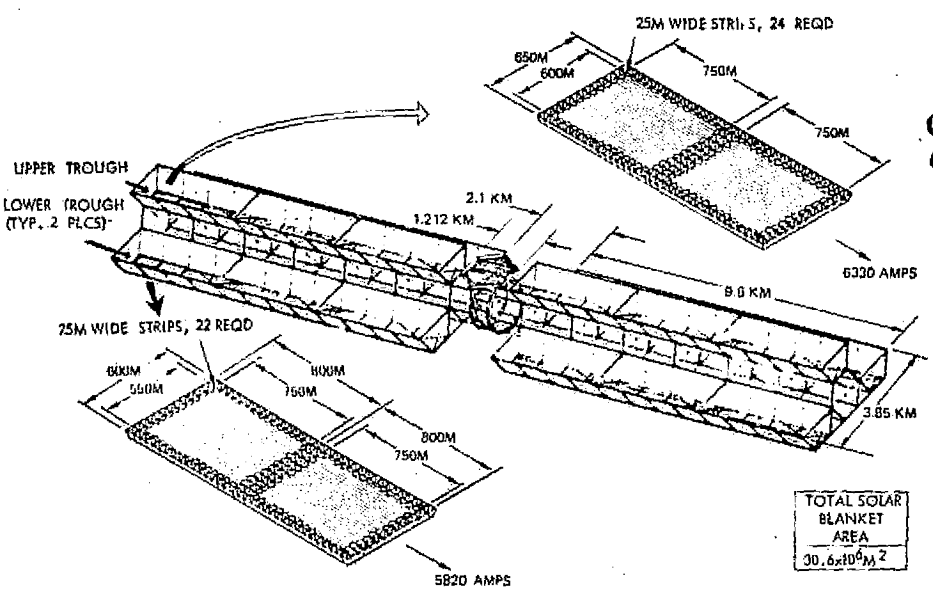


Figure 9.4-9. Solar Blanket Concept

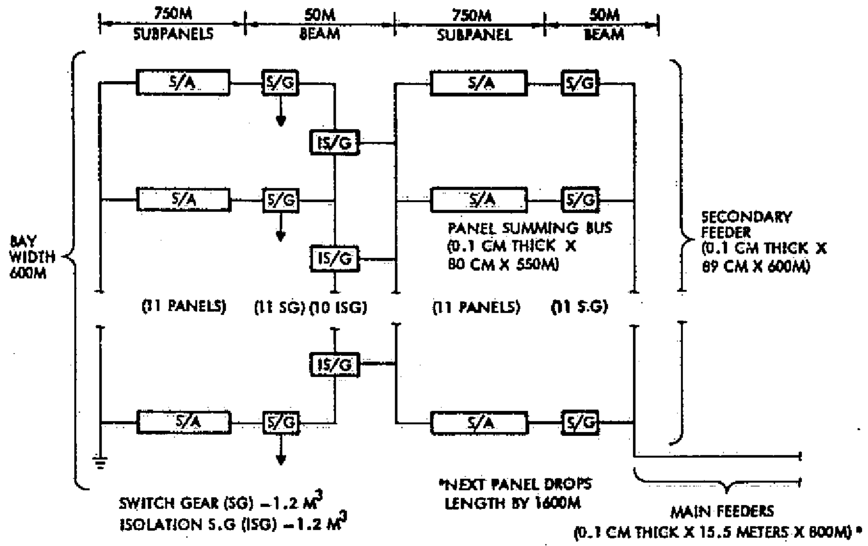


Figure 9.4-10. Two-Bay Solar Panel Wiring Schematic

50 m intervals along their length. [If the strips are exactly 25 m wide then an active length of 750 m per bay provides the required total satellite blanket area of 30.6×10^6 m^2 indicated in Figure 9.4-9. Flexibility exists within the dimensions of the primary structure to reduce the width of the strips somewhat (e.g., to 24 m) yet retain the total area requirement by increasing their active length (e.g., to 781 m). This flexibility has been retained in the event it becomes desirable to increase the separation between blanket strips along their longitudinal edges to improve installation and maintenance operations.]

The reader is referred to Figure 9.4-11 for a good visualization of the solar blanket installation equipment and operations. The solar blankets are installed in all three troughs simultaneously utilizing three sets of equipment and three crews. The solar cell blanket installation equipment is located on the construction facility in the bottom of each of the three troughs, indicated by Numbers 40, 41, and 42 on Figure 9.4-8. The blankets are packaged in 25 m wide rolls, each roll containing the 750 m length of active blanket to be installed in the 800 m long bays, the attached leading and trailing edge catenaries accounting for the remaining 25 m. The rolls are mounted on dispensing spindels across the width of each trough; 24 rolls wide in the upper (650 m wide) trough, and 22 rolls wide in each of the two lower (600 m wide) troughs. Construction-fixture-attached spindels for dispensing cables from rolls are located on the outside edges of the two outside blanket dispensers and between each of the internal blanket dispensers. There are 24 cable dispensers in the upper trough and 23 in each of the two lower troughs. One longitudinal catenary dispensing roll is mounted outside each of the two outside cable dispensers.

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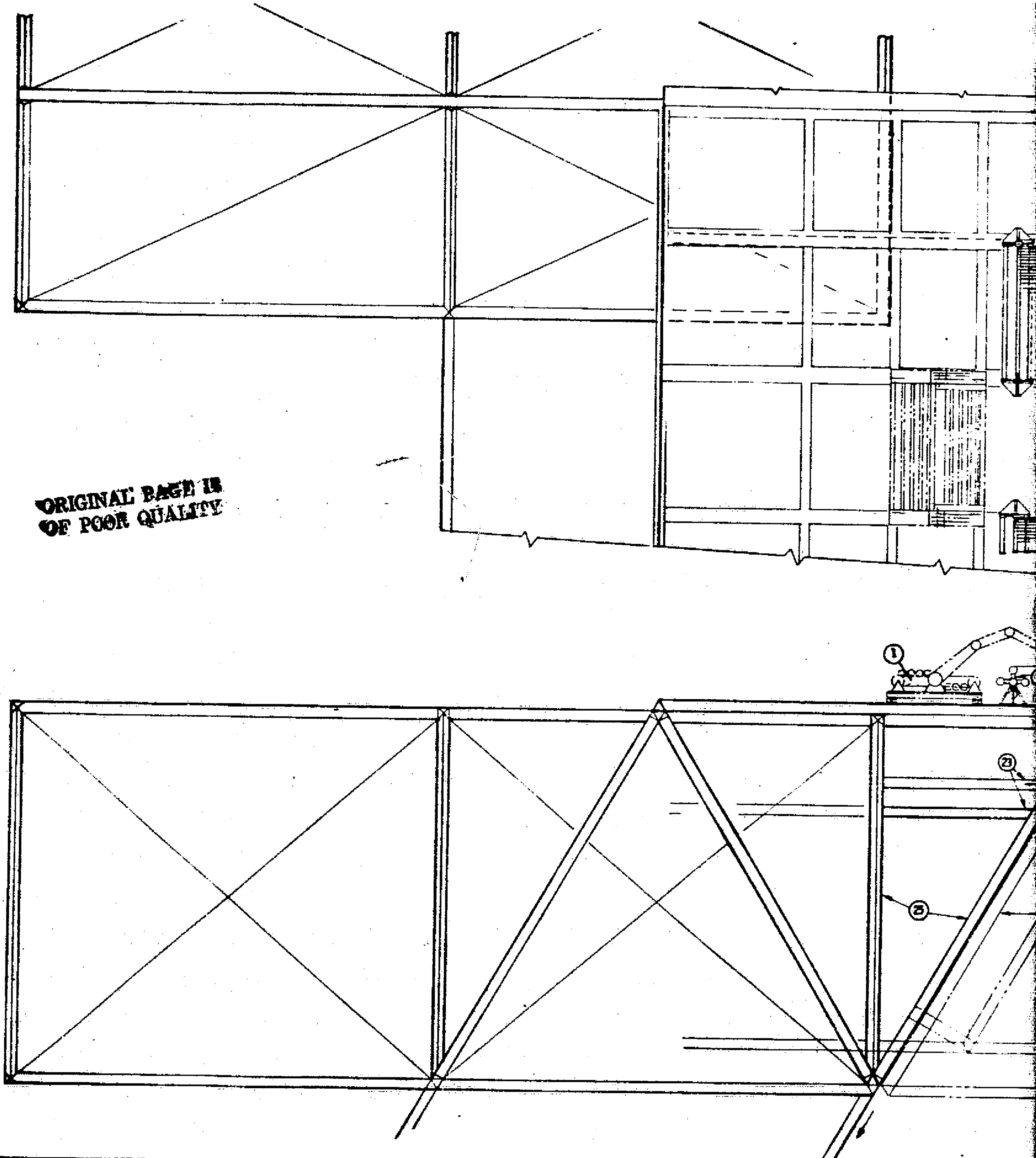

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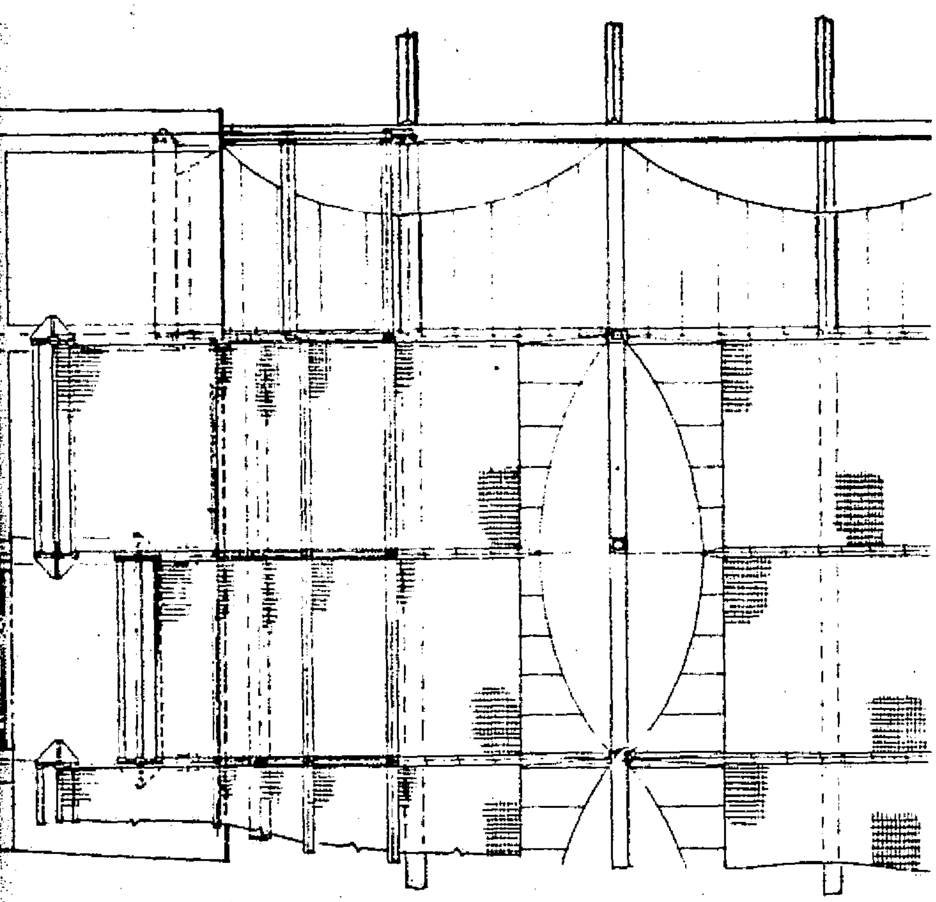

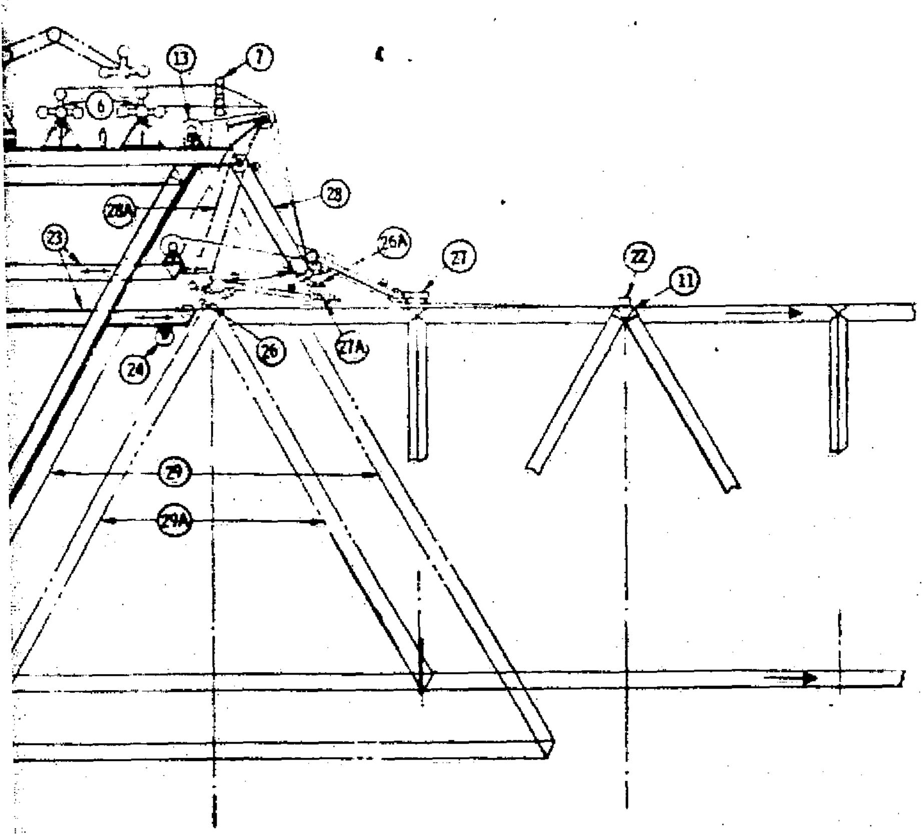

SOLAR CELL BLANKET INSTALLATION

1. S/A BLANKET ROLL TRANSPORTER - LOADER
2. S/A BLANKET ROLLS
3. USED ROLL CORES
4. BLANKET ROLL INSTALLER - REMOVER
5. TRANSPORTER TRACKS
6. S/A BLANKET DISPENSING SPINDEL QUAD*
7. BLANKET STRIP GUIDE ROLLERS
8. DEPLOYED SOLAR BLANKET STRIP
9. LEADING TRANSVERSE CATENARY
10. BLANKET STRIP - TRANSVERSE CATENARY JOINT LINE
11. UPPER VERTEX OF SATELLITE (50 M TRIBEAM GIRDER) CROSS BEAM
12. TRANSVERSE-CATENARY-TO-CROSS BEAM ATTACH POINT
13. LONGITUDINAL CABLE DISPENSER
14. LONGITUDINAL CABLE
15. LONGITUDINAL CATENARY DISPENSING SPINDEL
16. LONGITUDINAL CATENARY ROLL
17. UPPER VERTEX OF SATELLITE (50 M TRIBEAM GIRDER) LONGERON IN BOTTOM CORNER OF TROUGH
18. BLANKET-EDGE-TO-CABLE ATTACH MACHINE
19. BLANKET-EDGE-TO-LONGITUDINAL CATENARY ATTACH MACHINE
20. DEPLOYED LONGITUDINAL CATENARY
21. CATENARY-TO-LONGERON ATTACH POINT
22. SWITCH GEAR MOUNTED ON CROSS BEAM
23. RETRACTING PLATFORM FOR SWITCH GEAR AND SECONDARY FEEDER INSTALLATION
24. MAIN FEEDER DISPENSER
25. CONSTRUCTION FIXTURE
26. TRANSVERSE CATENARY-TO-CROSS BEAM ATTACH MACHINE IN ATTACH POSITION; 26A NON-ATTACH POSITION
27. LONGITUDINAL CATENARY-TO-LONGERON ATTACH MACHINE IN ATTACH POSITION; 27A NON-ATTACH POSITION
28. ATTACH EQUIPMENT TRANSLATING SUPPORT ARM
28A. TRANSLATING ARM IN CATENARY-TO-CROSS BEAM ATTACH POSITION
29. CROSS BEAM (50 M TRIBEAM GIRDER) FABRICATION FACILITY IN BEAM FABRICATION POSITION
29A. 50 M CROSS BEAM IN FABRICATION POSITION

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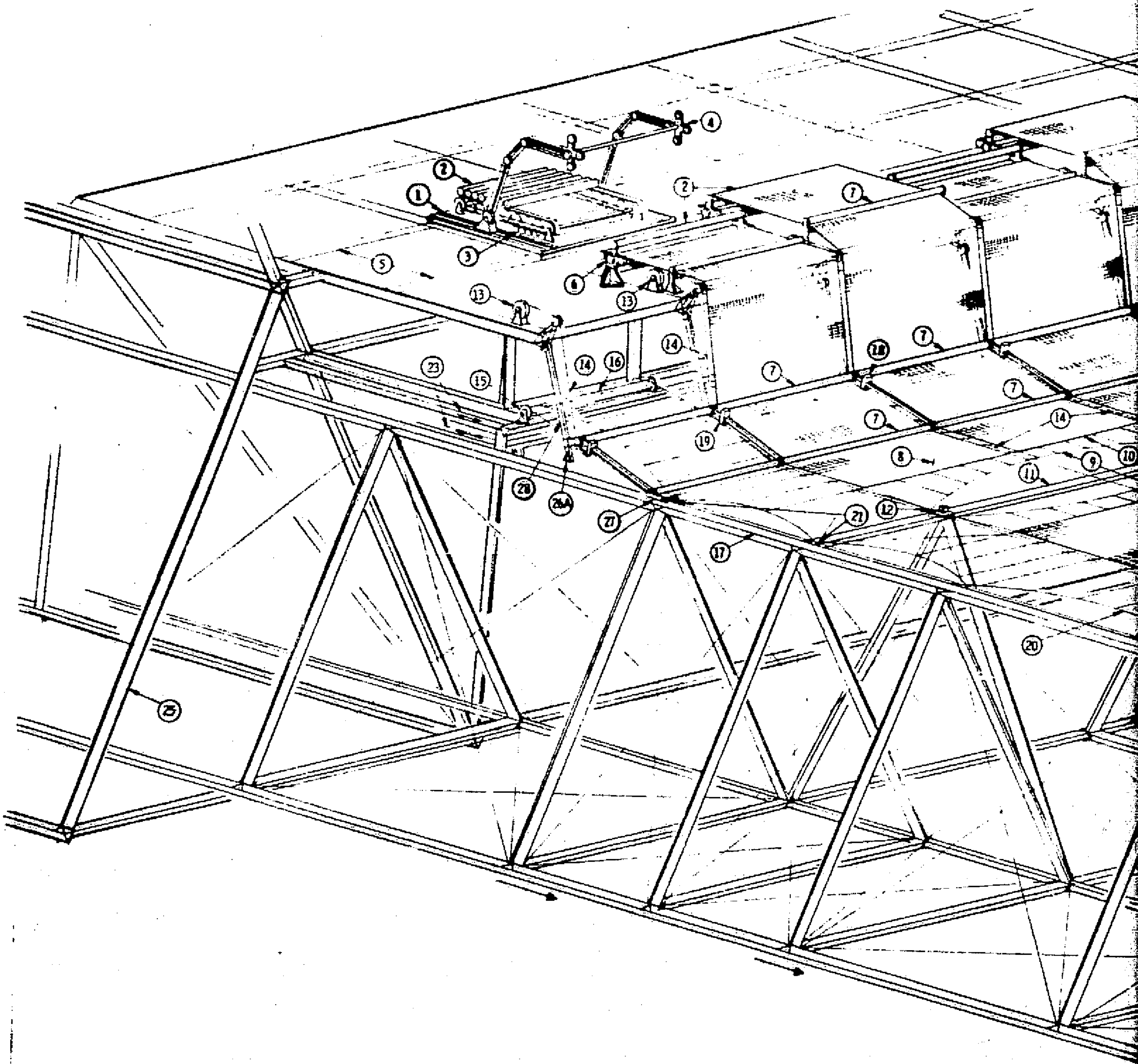

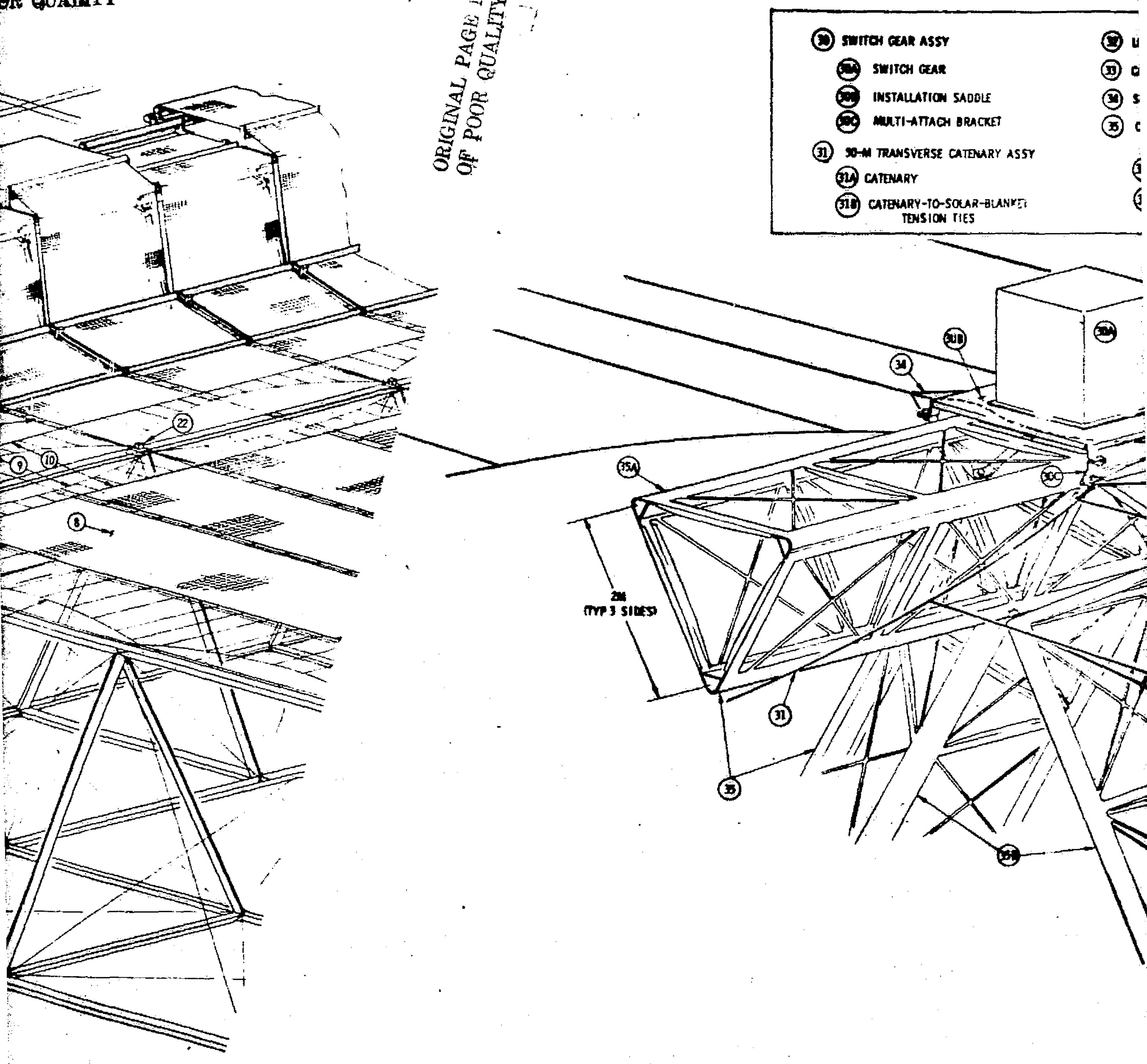

Figure 9.4-11. Solar Cell Bl

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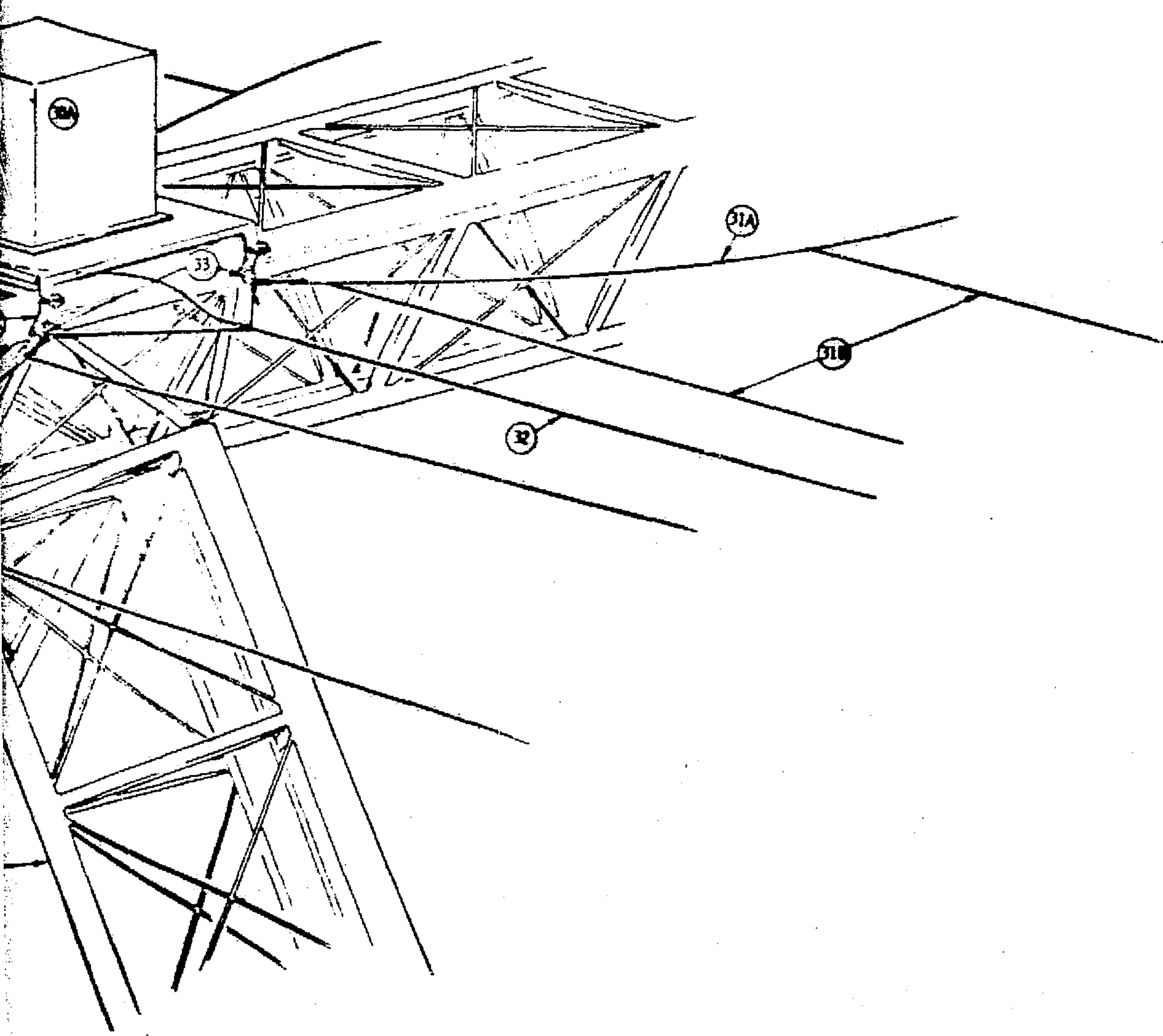
(32) LONGITUDINAL SOLAR BLANKET RESTRAINING CABLES
(33) CATENARY/SB CABLE ATTACH FITTINGS
(34) SB CABLE TENSIONING YOKE
(35) CROSS BEAM @ TROUGH BOTTOM (50-M TRI-BEAM GIRDER)
(35A) TOP CAP (2-M BASIC BEAM ELEMENT)
(35B) SIDE BEAM MEMBERS (50-M LONG)



Cell Blanket and Power Distribution System Trough Installation 9-27, 9-28

Solar blankets and catenaries are attached to the longitudinal cables by fold-over tabs which are applied by automatic fastening equipment. The fastening machines are mounted on the construction fixture downstream of the dispensing rolls. The fastening tabs are clip-fed into the machine as the cable and blanket edges thread through it.

The installation of the solar blankets involves the following sequence of operations: attaching the leading edge catenary of each solar blanket roll, the leading ends of the longitudinal cables, and the leading edges of the two longitudinal catenaries to the Nth cross beam just fabricated; paying out the solar blankets, longitudinal cables and two outside edge catenaries as the 800 meters of satellite longeron for the Nth bay are fabricated outward from the construction fixture; attaching, as the materials are played out, the two outside cables to their respective longitudinal catenaries, the two lonitudinal catenaries to their respective longerons, and the inside edges of adjacent blanket strips to their stabilizing cable; upon completion of the N+1 frame, attaching the trailing edge catenaries, the trailing edges of the longitudinal catenaries and the trailing ends of the longitudinal cables to frame N+1; tensioning the installation; and making the electrical connections to the switch gears, and to the secondary feeder where applicable.

The perspective drawing, Figure 9.4-12, illustrates the near-completion of the first three 800-meter bays showing part of the construction fixture, the local satellite structure, and the installed solar blankets and reflectors. A section of the outside reflector panels has been cut away to expose the solar cell blankets in the bottom of the trough.

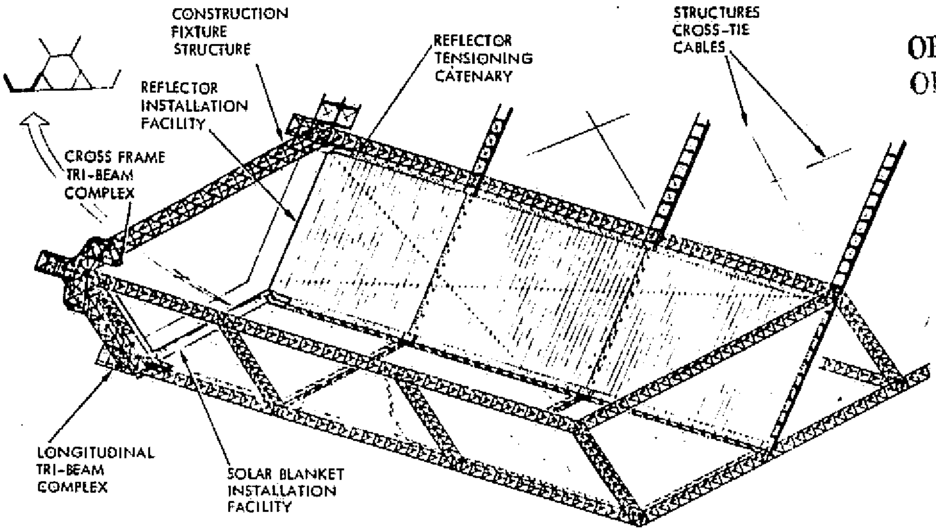


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Figure 9.4-12. Construction Perspective - Structures, Solar Blanket & Reflectors

Reflector Installation

The reflector panels, measuring 600-m x 800-m, are pleated at 25-m intervals to produce an accordian type fold as shown in Figure 9.4-13.

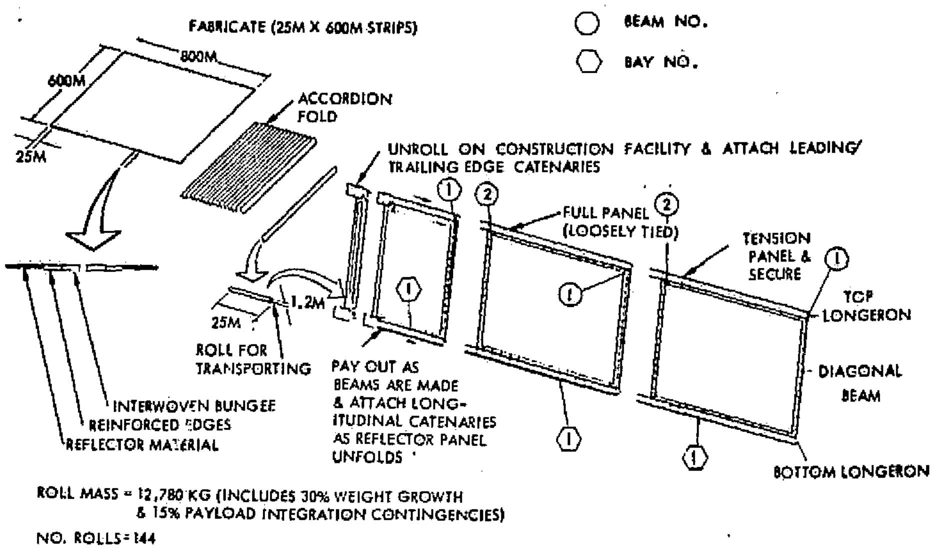

Figure 9.4-13. Reflector Packaging & Installation Concept

They are then rolled along the plane of the end pleat into a roll 25-m long and 1.2-m diameter which is the configuration for transporting into orbit.

When installed, each reflector panel is suspended within the 800-m bay by longitudinal catenaries attached to the upper and lower longerons and by leading and trailing edge catenaries attached to the forward and aft diagonal members of the transverse frames. The cantenaries are attached to the trailing and leading diagonal transverse beams and to the longerons as shown in Figure 9.4-13. Two panels are required for each 800-m bay of each trough or a total of 144 panels for the entire satellite.

To install the reflector, the roll is first unfurled into a 25-m x 600-m configuration along the diagonal of the construction fixture (reference items 34-39, Figure 9.4-8), and positioned in a dispensing device which permits deployment without crinkling. The leading edge of the reflector is fastened to the leading edge catenary. This catenary is secured to the diagonal member of the transverse beam (Beam 1 in Figure 9.4-13) which has already been fabricated. As the longeron for Bay 1 is fabricated, Frame No. 1 moves away from the construction fixture, taking the leading edge of the reflector with it and the rest of the reflector panel is payed out from the dispensing device on the facility. As the reflector panels are payed out, the longitudinal catenaries are attached to the panels, and in turn the catenaries are attached to the longerons. These operations and the equipment used are identical to those described for the solar blankets attachment to the longerons.

The longitudinal catenaries are fastened to the longerons as they emerge from the beam machine, utilizing either manual EVA or a manned manipulator module. The trailing cable and catenary are then attached to the reflector trailing edge. After Beam No. 1 has advanced 800 m to its final position and Beam No. 2 has been installed in place, the trailing edge catenary is fastened to Beam No. 2 in the same manner as the attachment for Beam No. 1. The cables

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are then tensioned and secured, which completes the panel installation operations for the bay. The same procedure is then followed in subsequent bays.

Switch Gear Installation

Switch gear for voltage regulation and isolation control of the solar blankets are installed on the cross beams at the bottom of the troughs. As previously discussed, the two-bay wiring schematic, Figure 9.4-10, shows that switch gears are mounted on each of the cross beams (starting with frame No. 2, Figure 9.4-2) that form the bottom of the solar converter troughs. Each switch gear has a volume of about 1.6 m^3 and weighs about 240 kg. Twenty-two units are mounted on even numbered frames and 11 units on odd numbered frames. They are stowed in tiered containers located at the proper installation station on the construction fixture with attached saddle type mounting straps so that the assemblies can be placed in position over the vertex of the cross beam, fastened in place, and the electrical junctions effected with simple movements. Remote manual execution of this installation is envisioned using two 2-crew-member pressurized (shirt sleeve) modules with manipulator arms in each trough. The module, operating on a track attached to the construction fixture has the capability to position itself over the cross beam at any location across the width of the trough. Installation can be accomplished in an 8-hour shift and allow 45 minutes per switch gear at the maximum density (21 units) beams.

Main Feeder Installation

The main feeders which conduct dc electrical energy from the solar cell blankets in the satellite wings to the central section near the slip rings are routed longitudinally on the undersides of the 50-m cross beam in the area beneath the longitudinal solar blanket catenaries. The positive (+) and negative (-) are located on the opposite sides of the trough for maximum separation. These feeders are thin-gauge (0.001-m) aluminum sheet stock, which is delivered to the construction fixture in rolls of six different widths ranging from 0.12 to 1.57 m. The longest feeder starts at completion of the second bay of solar blanket installation and, at completion of each two succeeding bays, another feeder is added. A total of 12 feeders [6 position (+) and 6 negative (-)] tie into the summing buses at the end of each trough, near the center of the satellite.

The length of the feeder on a single roll is either 800 m or 1600 m (one- or two-bay length) depending on the width; this is to facilitate handling by limiting roll weights to 5000 kg. The dispensing spindles for the main feeder rolls are mounted on the lower level of the construction fixture (Item 24 of Figure 9.4-11) to allow deployment on the underside of the satellite cross beams.

Installation is initiated by attaching the leading edges of the longest feeders to the trough cross beams of satellite frame No. 2 (Figure 9.4-2) which has just been completed and effecting the junction with the secondary feeder there. As the longerons for satellite bay No. 3 are fabricated, pushing frame No. 2 away from the fixture, the feeders are payed out from the dispensing rolls. Upon completion of frame No. 3 the trailing edges of the bay No. 3 feeders are spliced onto the leading edges of the feeder rolls for bay No. 4. This process is repeated until the wing is completed. Every second bay another pair of feeders is added to carry the current from those bays to the center.

Microwave Antenna Element Installation

The microwave antenna is constructed concurrently with the first wing and then translated and rotated into its final position on the rotary joint as shown in the sequences of Figures 9.4-14 and 9.4-14A. The work fixture for constructing the antenna frame is shown in Figure 9.4-14. The hexagonal portion serves as a jig for the antenna; the remainder of the fixture provides a jig for the beam machines which produce the satellite main structure (longerons are fabricated in the directions indicated by the arrows). The fixture provides capability for translation and rotating the frame into final position as in the sequence of Figure 9.4-14A.

Panel A of Figure 9.4-14 shows the start of antenna frame construction. The 50-m tribeam machine complexes are utilized for all six sides. Initially the corners are constructed, followed by the connecting beams.

After the antenna frame has been completed, catenary cables and the tension (or suspension) web composed of composite cables is installed. Panel B of Figure 9.4-14 shows the completed installation. Figure 9.4-15 shows details of the cable arrangement which serves as a mounting for the mechanical modules. Two vehicles which are able to traverse the perimeter of the antenna frame by means of a track or cable attached to the antenna frame are used to deploy the tension web cables. The vehicles are maneuvered into positions opposite to each other. Each cable is deployed from one vehicle to the other by means of a closed loop leader similar to clotheslines which is attached between the two vehicles prior to their moving into position. The deployed cable is attached to the catenary by a manned vehicle equipped with manipulators. Conceptual layouts for these vehicles are included in the proposed tasks for the follow-on study.

After installation of the suspension web but before attachment of the antenna to the rotary joint, the mechanical modules must be installed into the antenna frame and secured to the suspension webs. Two considerations identify this installation as the critical path in the satellite construction schedule: cargo density considerations dictate the assembly of a very large number of rf elements (approximately 136000 power modules and 777 mechanical modules) at the construction base. Because of MW taper requirements, 10 different mechanical module configurations are required. Each module is separately installed to provide specific patterns across the face of the antenna. Figure 9.4-16 shows the relationship of typical power modules, subarrays, and mechanical modules and illustrates the pattern effect on the antenna suspension web. Two actions have been taken to relieve this criticality. The antenna fabrication schedule has been maximized (66 days) by innovative design of the satellite construction fixture (as described in Section 9.4 "Nth Satellite Construction Schedule"); and a concept for assembly and

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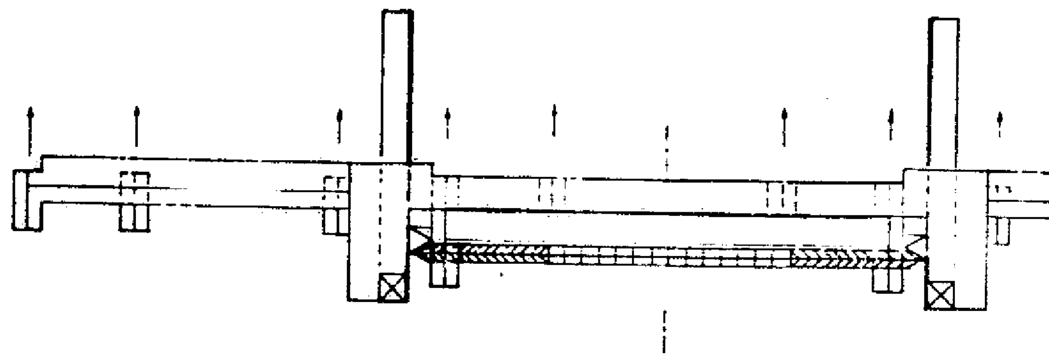

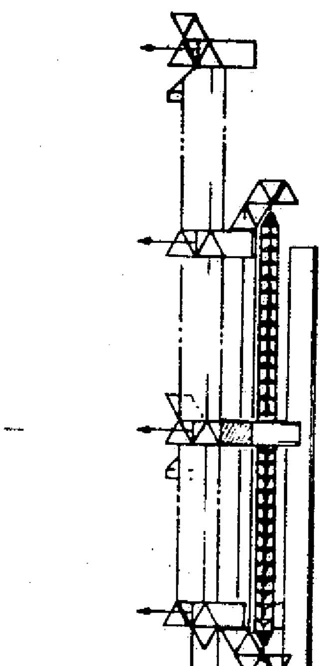

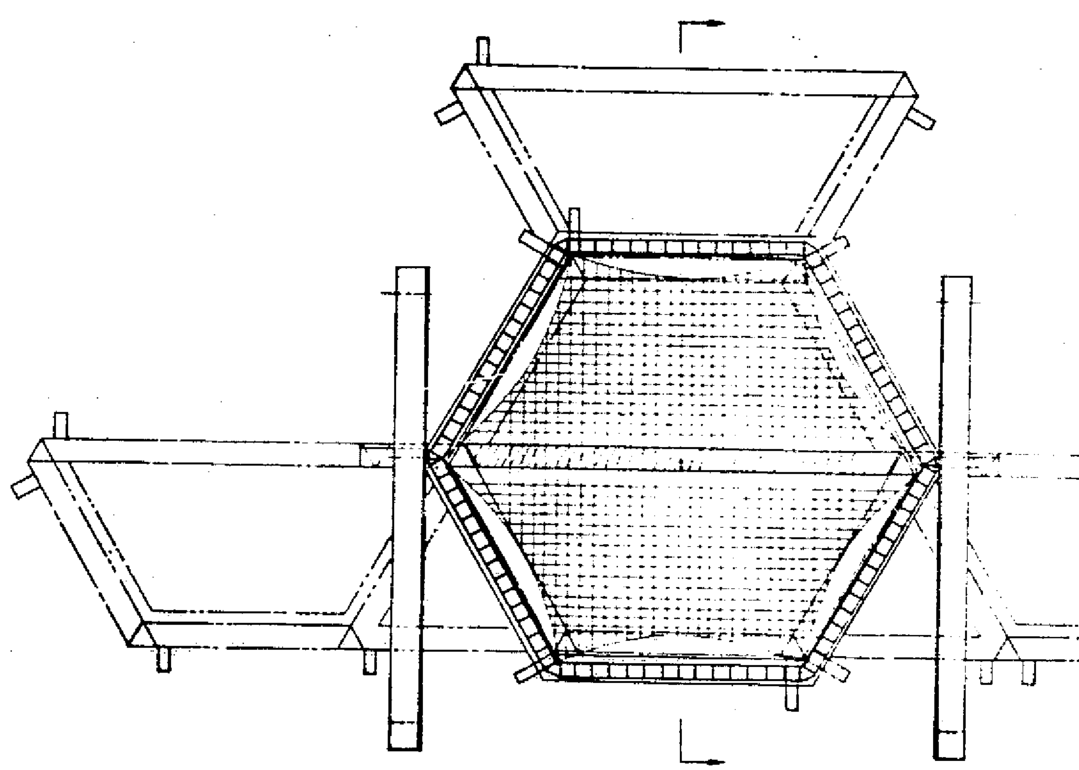

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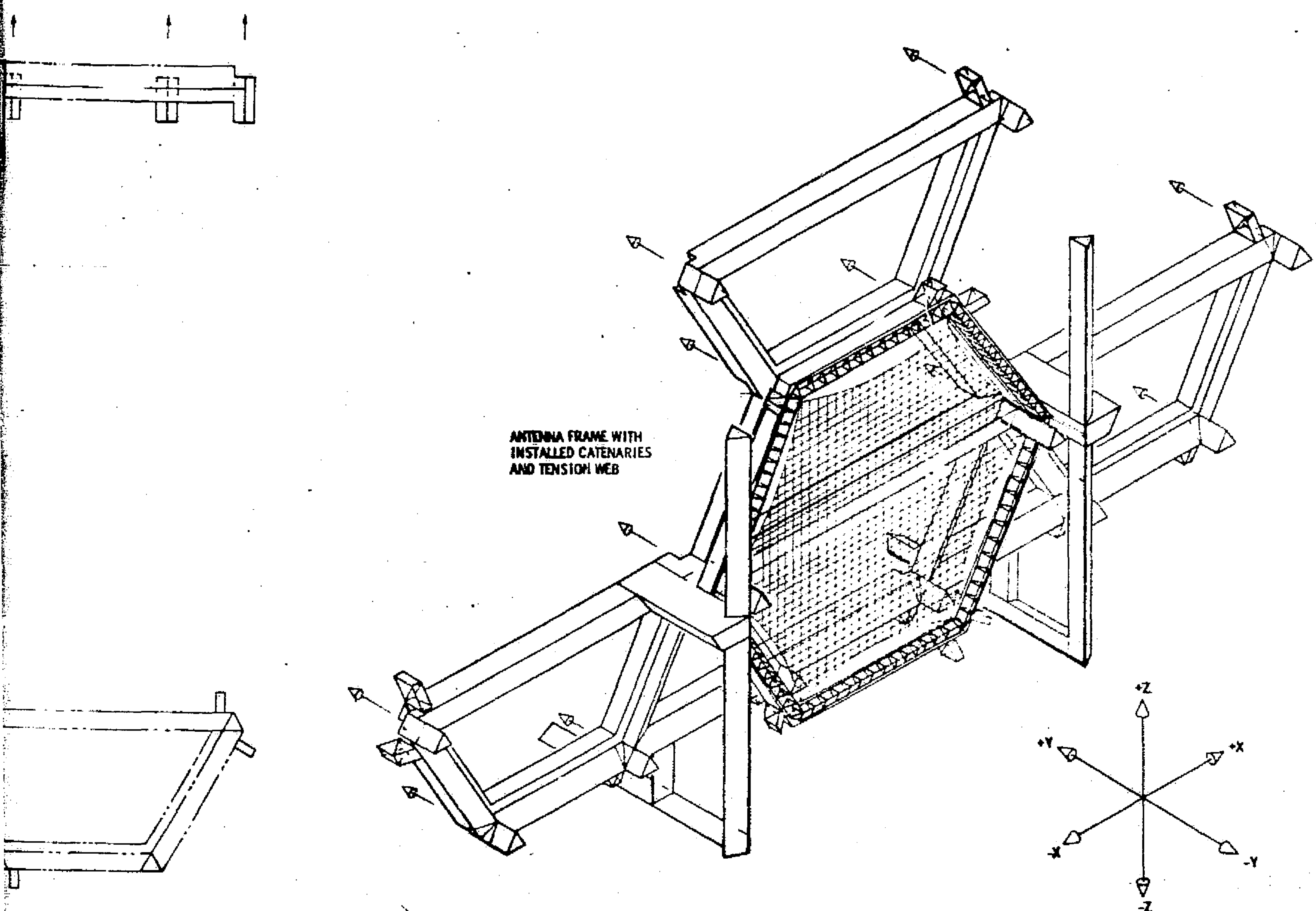

PANEL B

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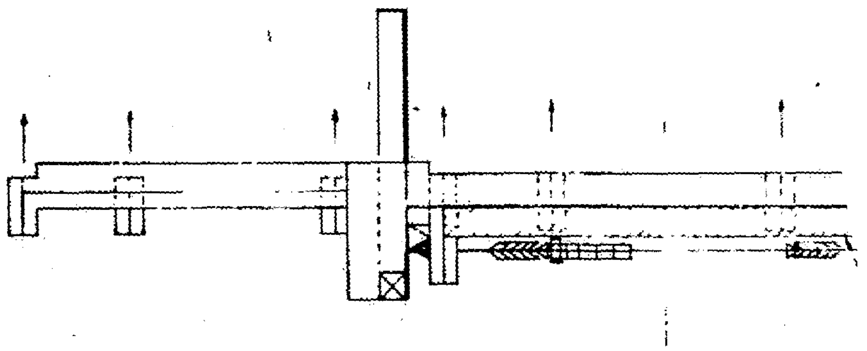

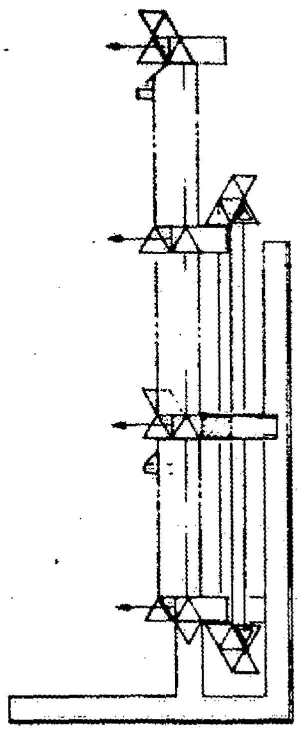

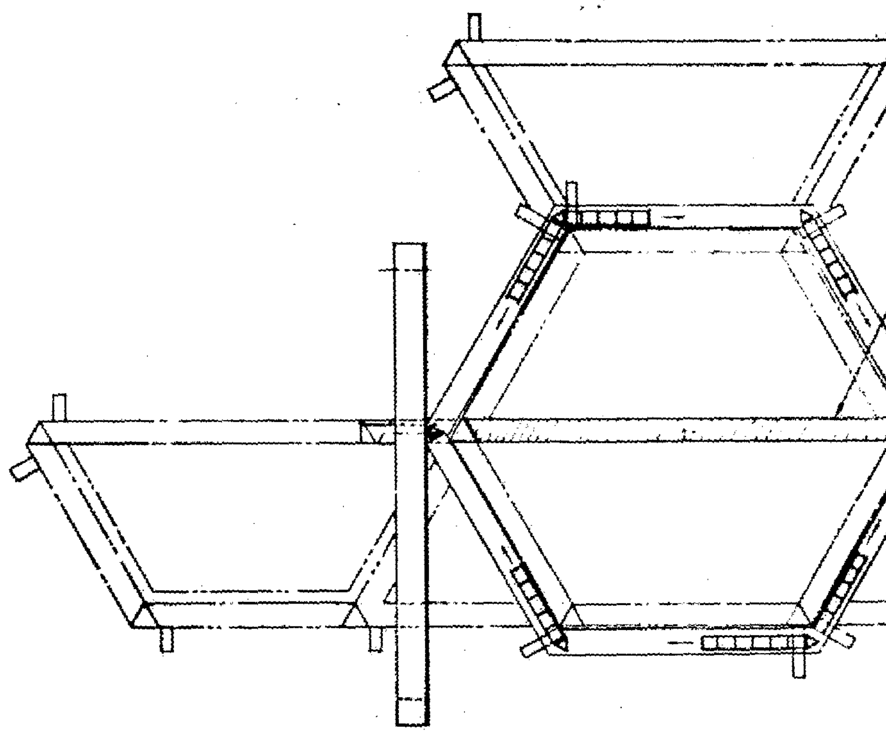

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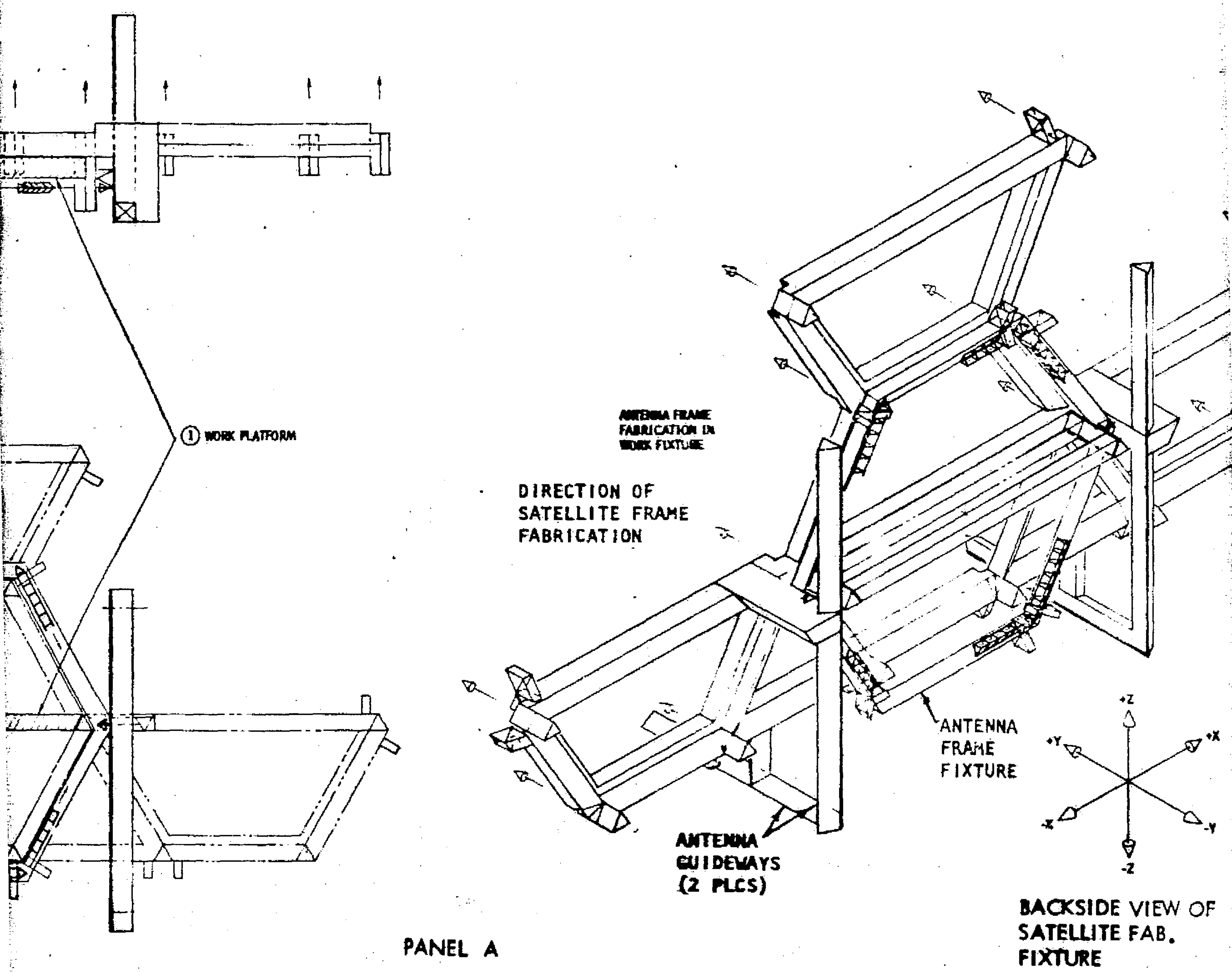

Figure. 9.4-14. Antenna Construction and Installation Scenario

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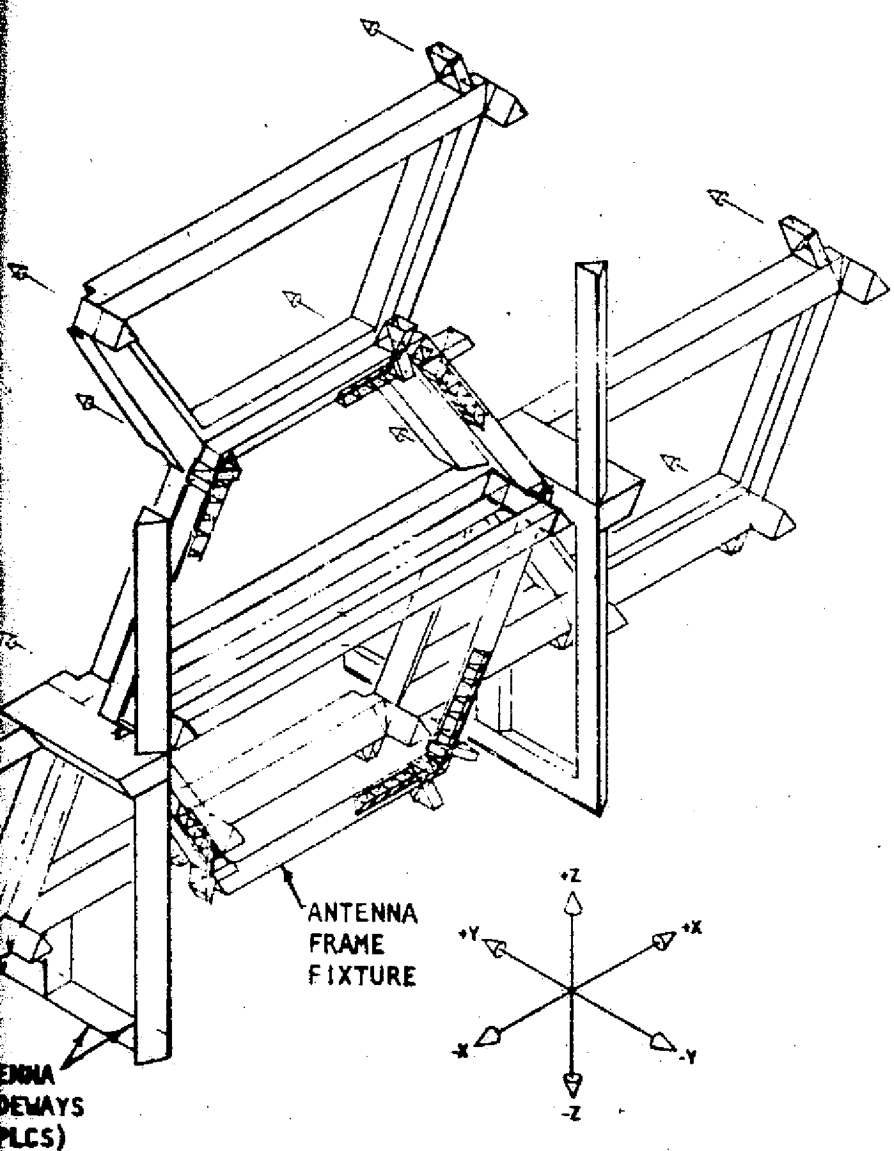
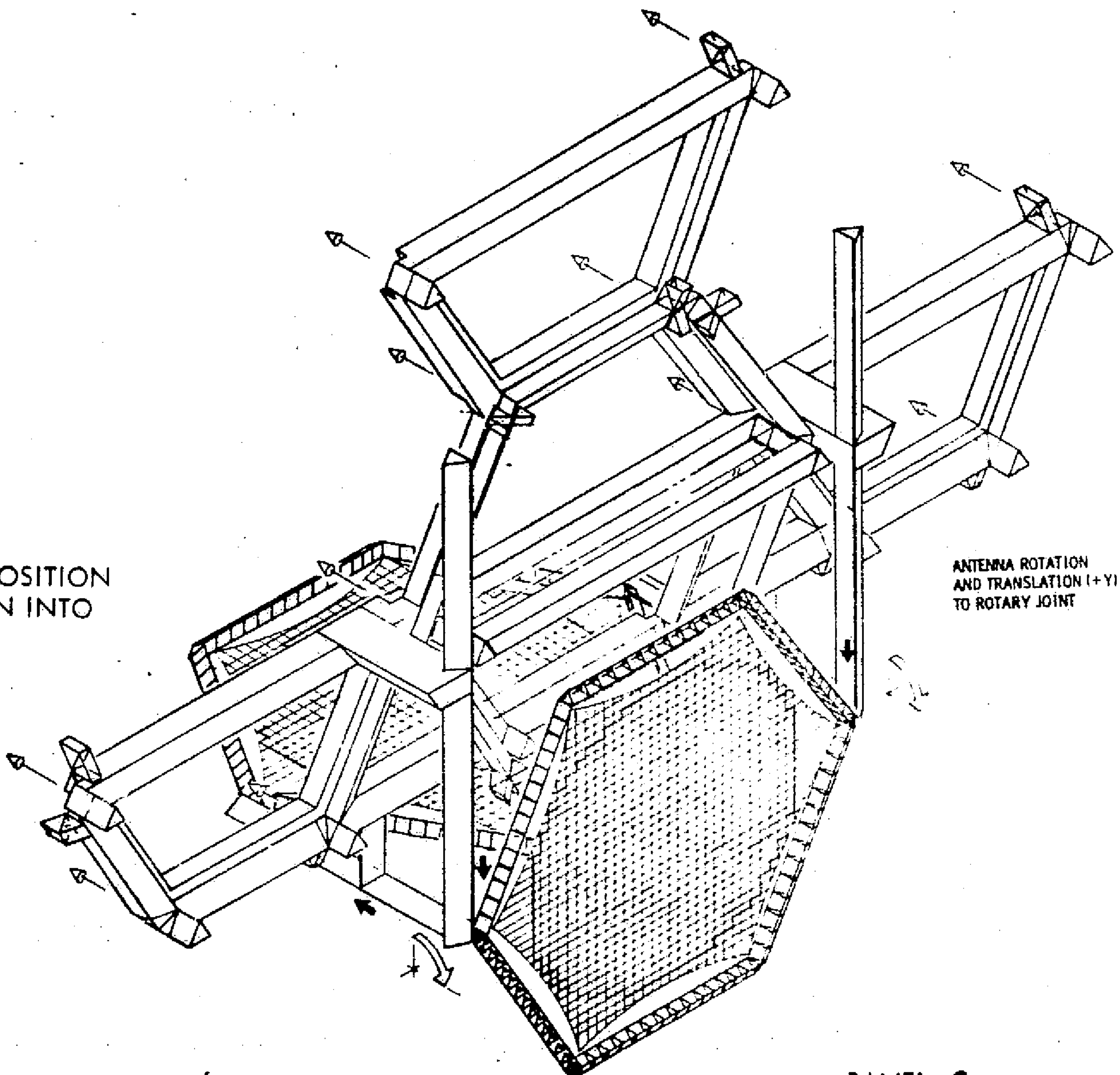


BACKSIDE VIEW OF SATELLITE FAB. FIXTURE

ANTENNA FINAL POSITION
FOR INSTALLATION INTO
ROTARY JOINT



PANEL C

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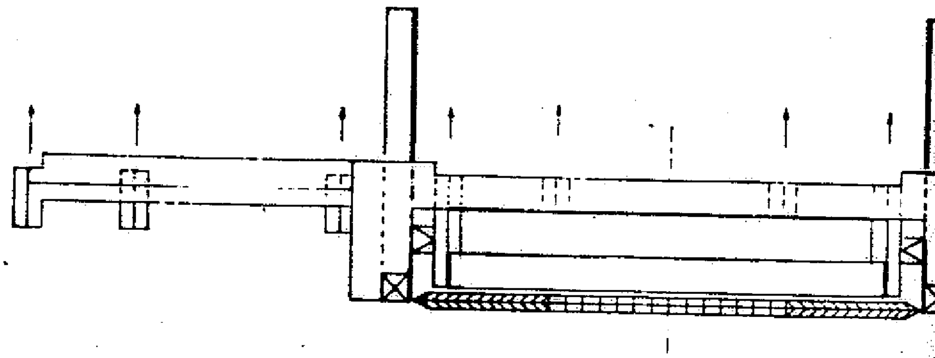

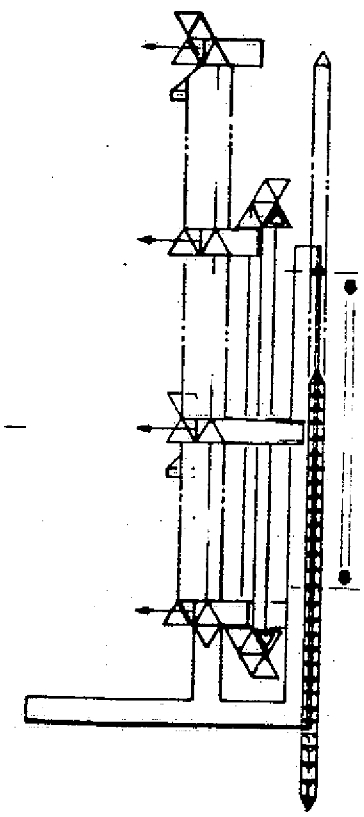

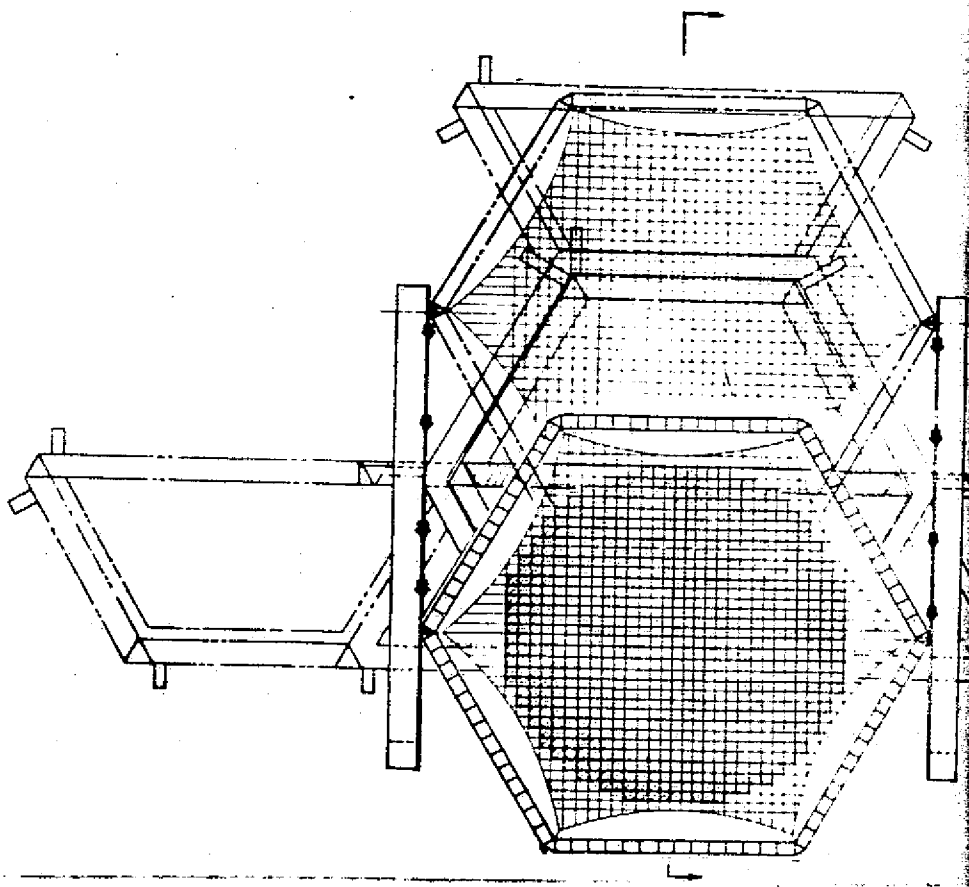

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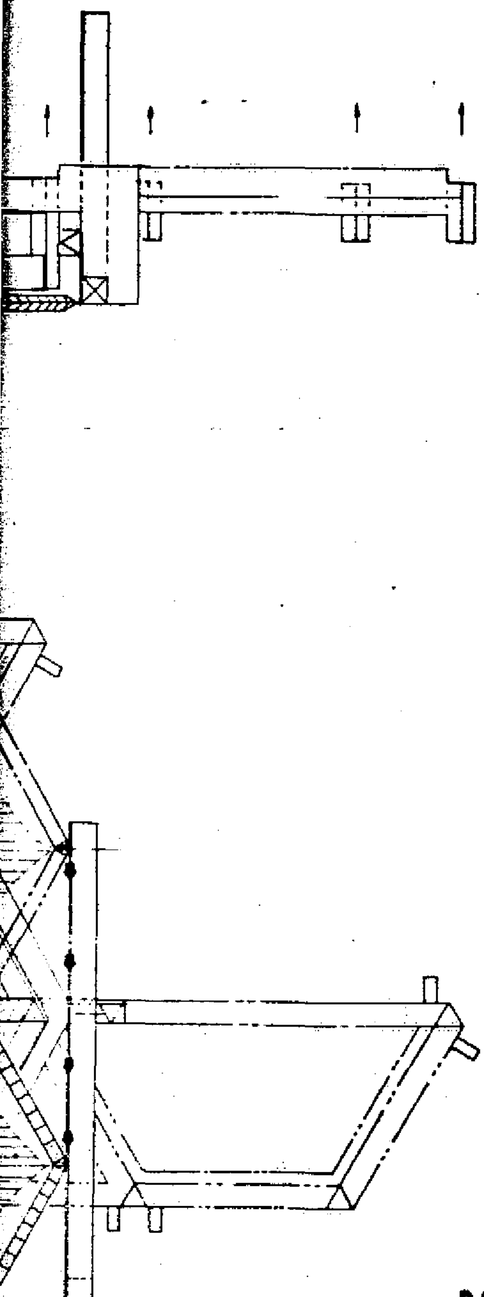

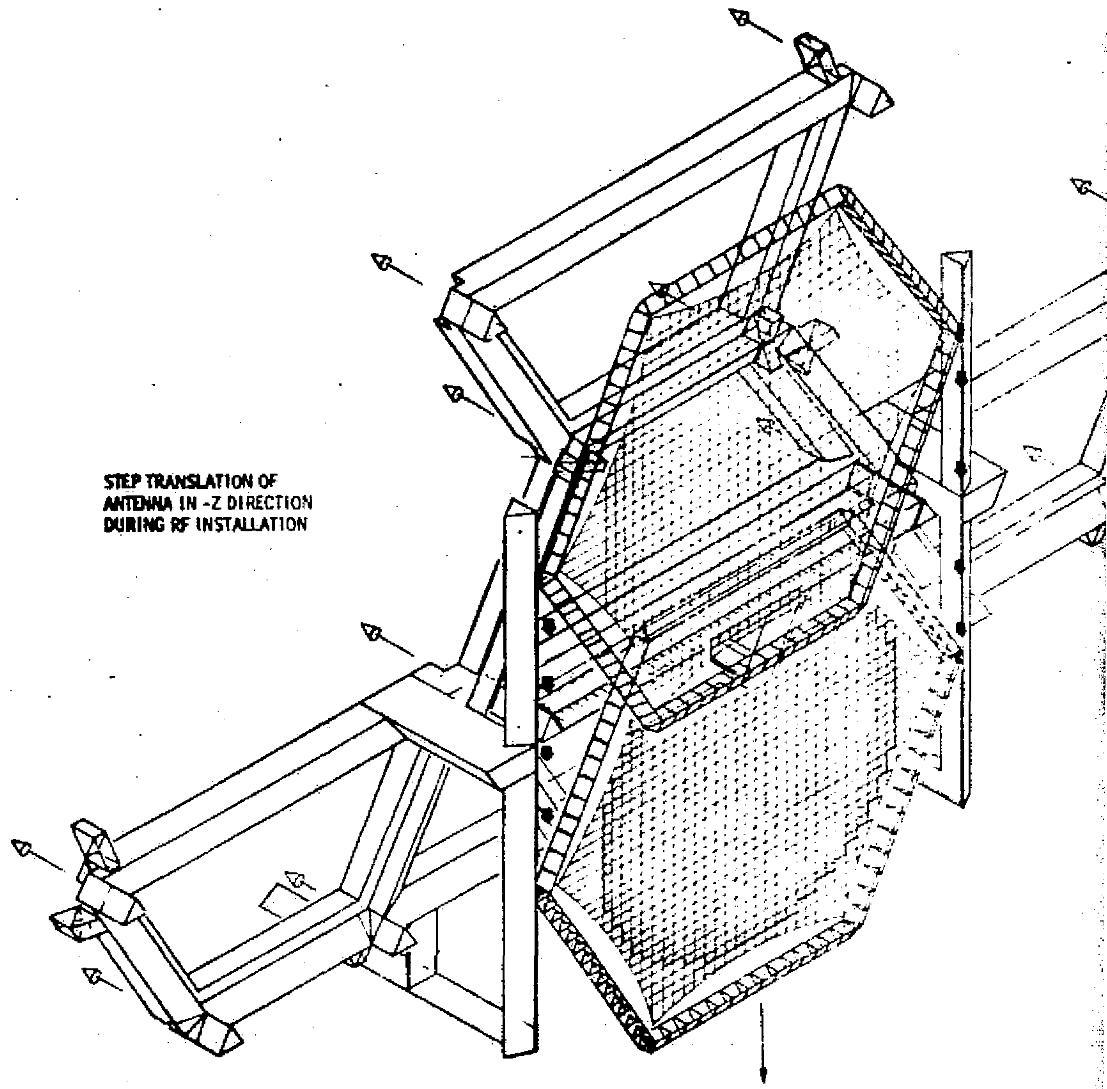

PANEL 8

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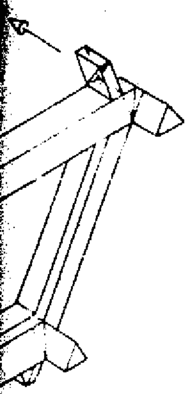

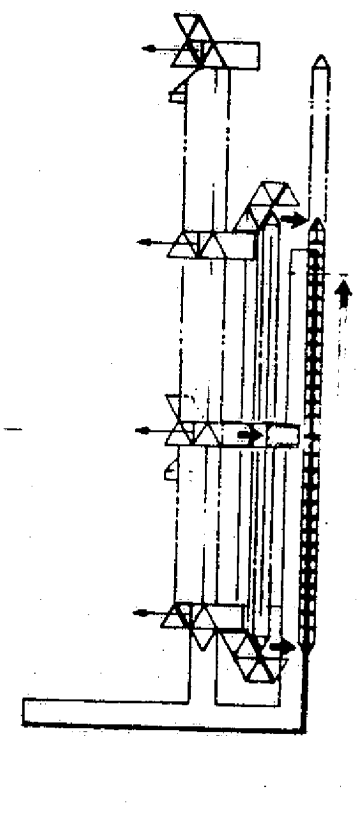

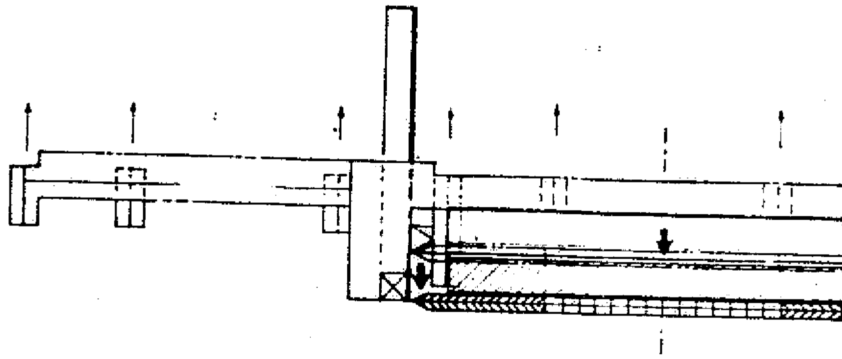

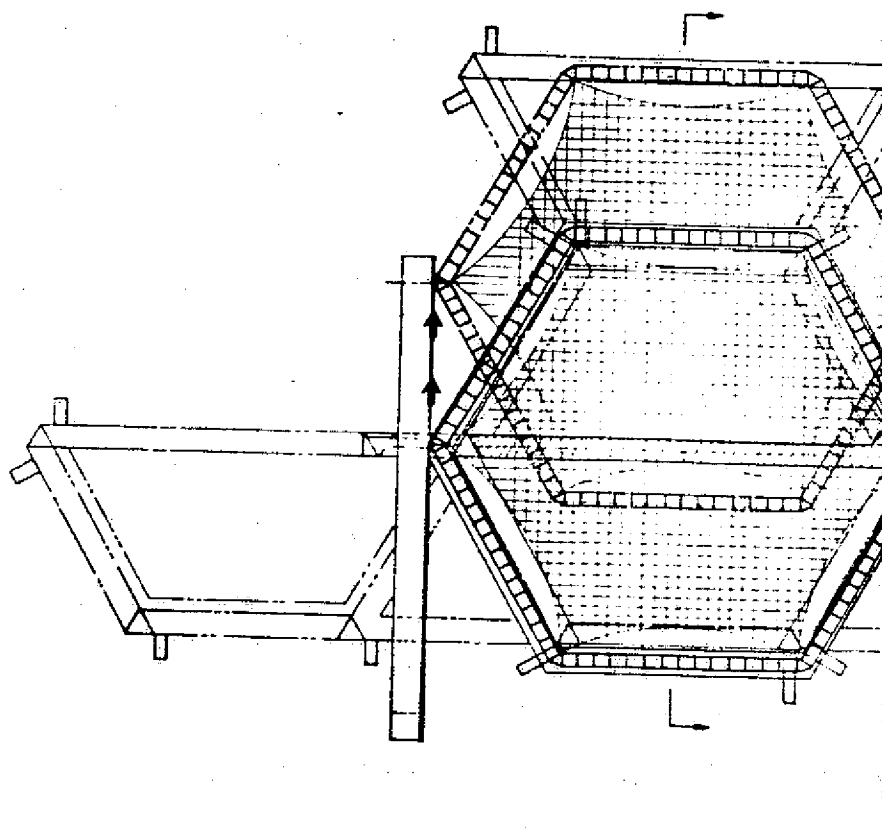

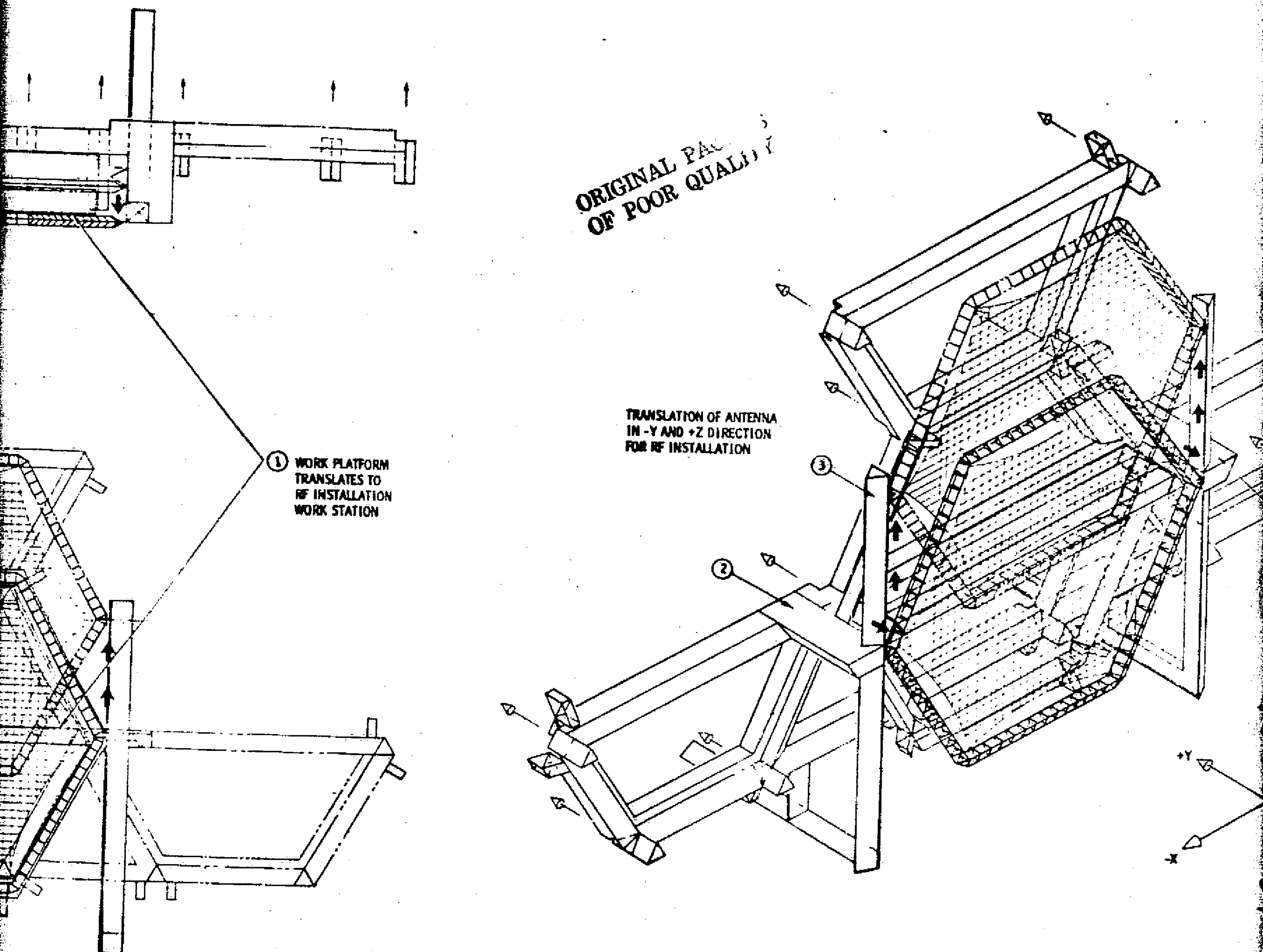

PANEL A

Figure 9.4-14A. Antenna Translation Scenario

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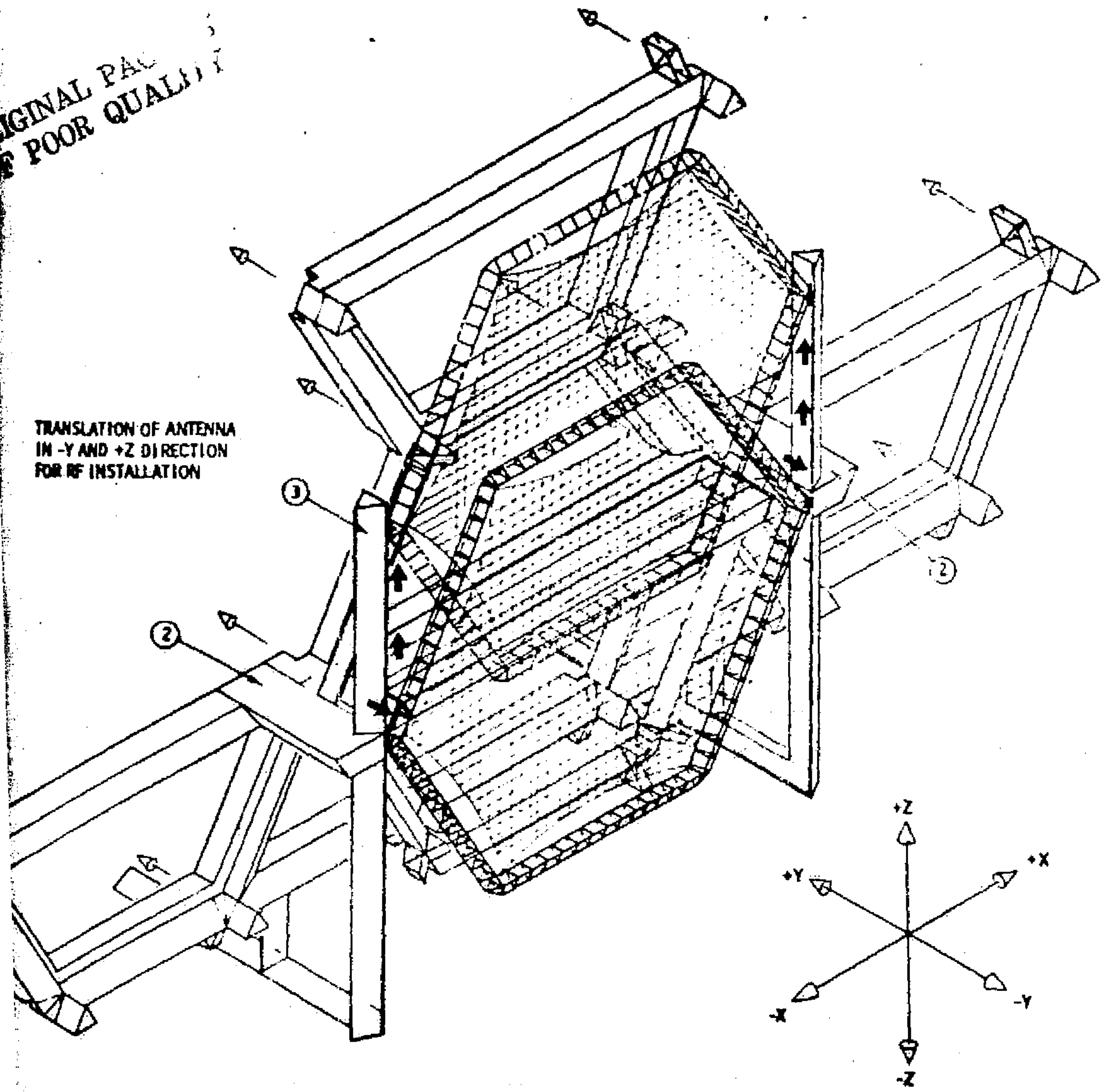


igure 9.4-14A. Antenna Translation Scenario

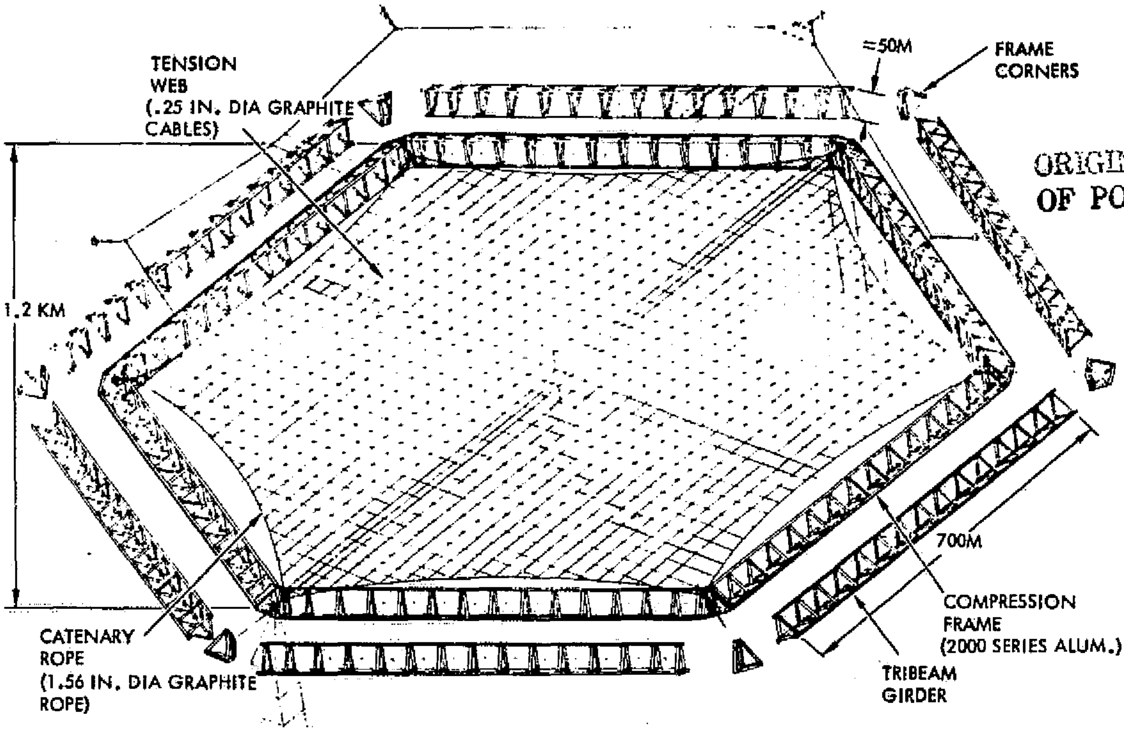


Figure 9.4-15. Spacecraft Antenna Structure

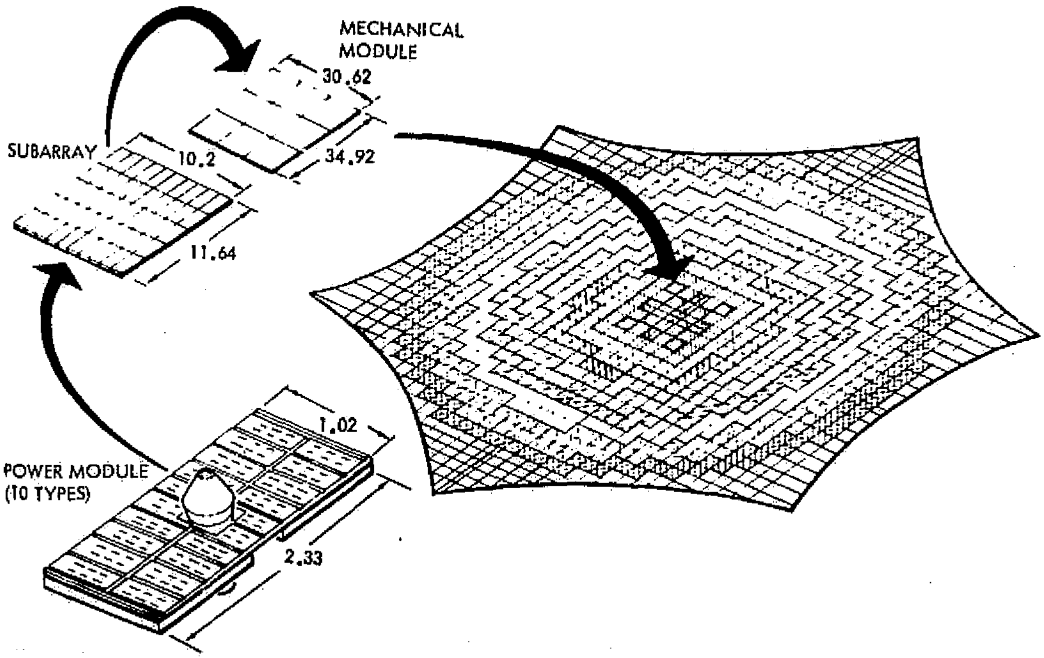


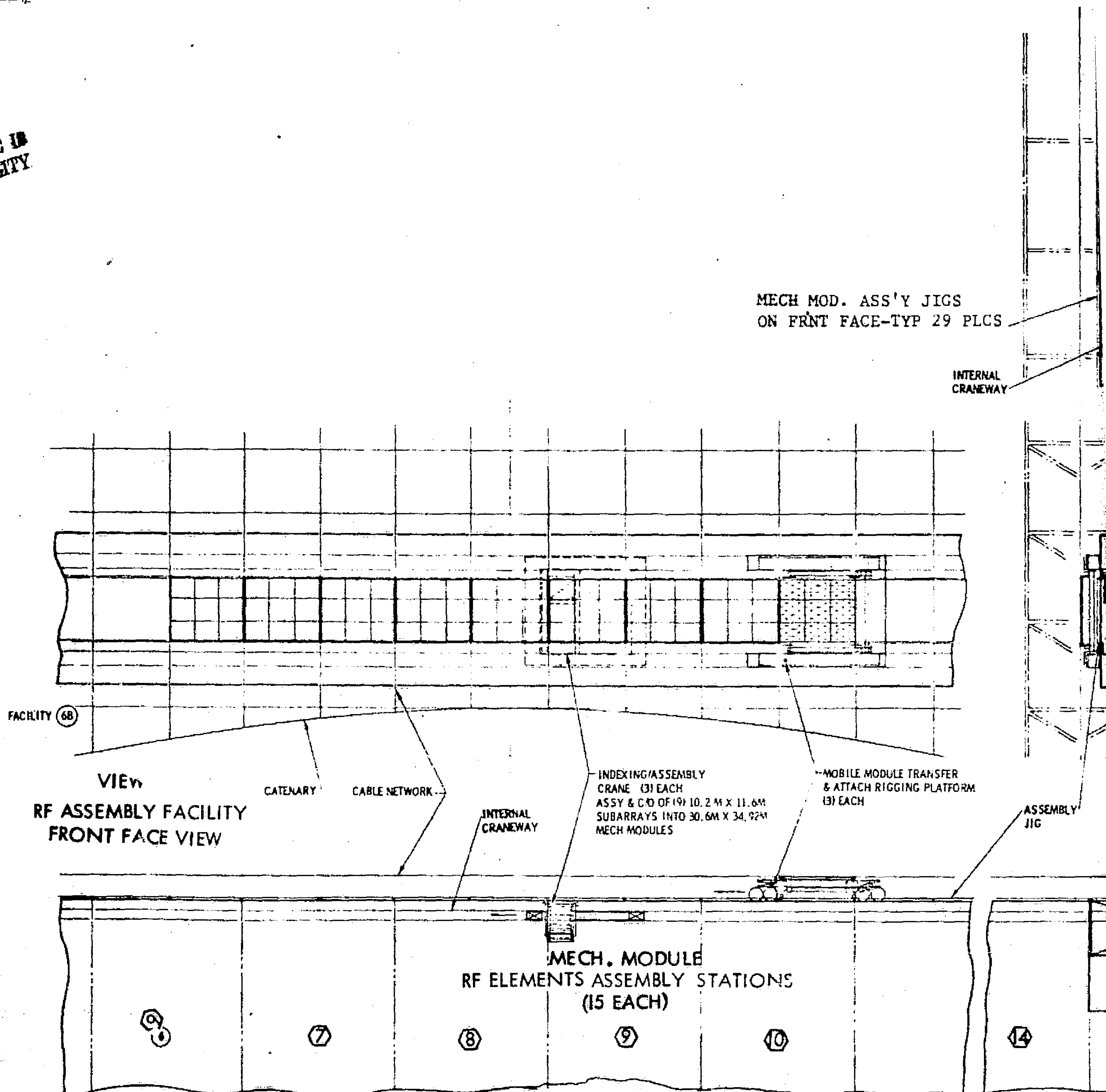
Figure 9.4-16. Microwave Transmission System
- Satellite Antenna

installation of the rf elements at a rate which supports that schedule with ample margin has been conceived as described in the following paragraphs. This concept maximizes automated handling, assembly and installation operations and minimizes material transfer distances.

The rf assembly and installation facilities and sequence of operations are depicted in Figure 9.4-17. (All subsequent references in this discussion will be to Figure 9.4-17 unless stated otherwise.) The antenna waveguide subarray panels, klystrons, and electronic modules are delivered to the rf assembly and installation facility, Item 6B, in the packaged configurations described later in Figure 9.7-1. A matrix identifying installation location on the antenna suspension web for each mechanical module by file and row number is given in View A. The rf assembly and installation facility, 6B, is behind the antenna in View A. View B, looking down from the top edge of the antenna, shows that the rf facility is supported at each end from the satellite fabrication fixture structure, and that it spans the full width of the antenna. The numbers across the top surface of the facility indicate the locations of 15 identical work stations for processing the incoming rf elements preparatory to assembly into the 30.6-m x 34.92-m mechanical module configurations. Each of the 15 work stations assemble modules in the sequence required to support installation of specified file numbers. Waveguide subarray panels, klystrons, and electronic modules are sorted at the cargo receiving area (Item 4 on the SCB perspective), and recomposed into packages which support the specific demands of the various work stations. Here the critical elements are the waveguide subarray panels, because of the non-uniform variation in module power density configuration along files and rows. The repackaged rf elements are loaded onto a transporter (identified in View B') which delivers them to the designated work stations where they are placed in their respective holding areas.

Processing at the work stations is described in View B' and C'. The waveguide subarray panels are unfolded to their operational surface dimensions of approximately 11-m × 10-m (the panels are about 0.26-m thick). They are then passed through the mechanical checkout station and klystron indexing/assembly station where the klystrons are automatically installed. The assembly of subarray and klystrons move to the electrical station where the electronic control boxes are installed, electrical connections are made, and the assembly is functionally checked out. Nine assemblies of identical configurations are loaded into the elevator magazine which transfers them (View C') down to the mechanical module indexing/assembly crane which operates on a craneway behind the front face of the facility (Views C', E and F). Jigs for assembly of the mechanical modules are located in the front face of the facility (Views C', E and F), one jig in line with each of the 29 files on the antenna. Three cranes service the 29 jigs. Each mechanical module is an assembly of 9 subarrays. The crane indexes the magazine to each of the 9 sections of the jig, installing a subarray in each, automatically making the connections and testing for mechanical module integrity.

Three mobile module transfer and attach units travel the external front face of the rf facility with access to each module jig and to each antenna file (Views E and F). The antenna frame with its module support web installed translates downward in its guideways, (6C) bringing each row sequentially in front of the rf facility. The module transfer units remove the completed



VIEW E RF ASSEMBLY FACILITY BOTTOM VIEW

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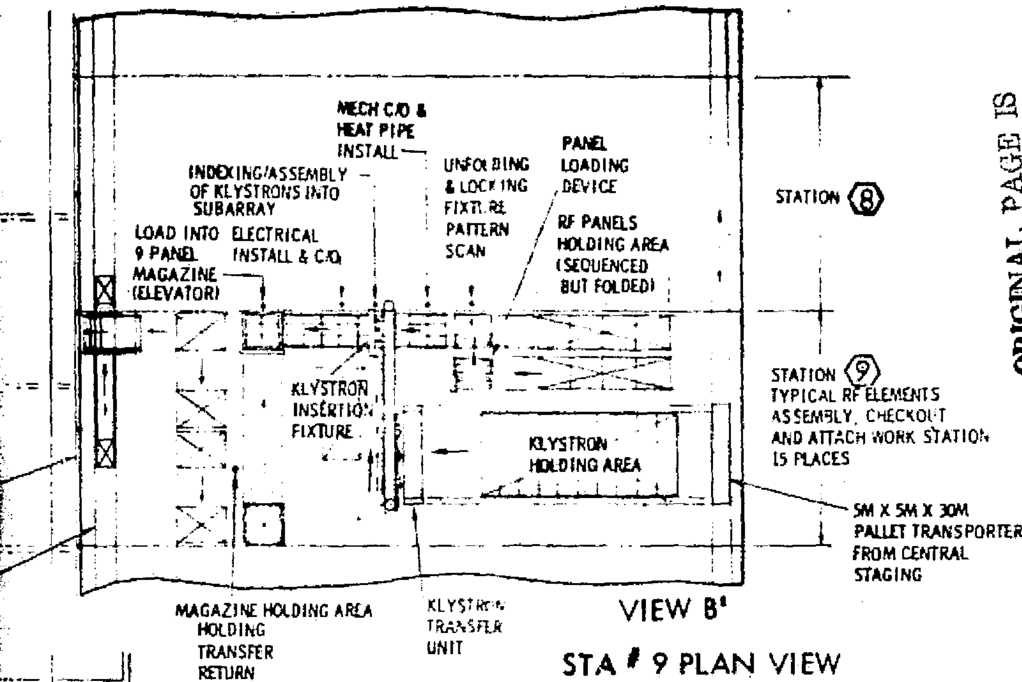
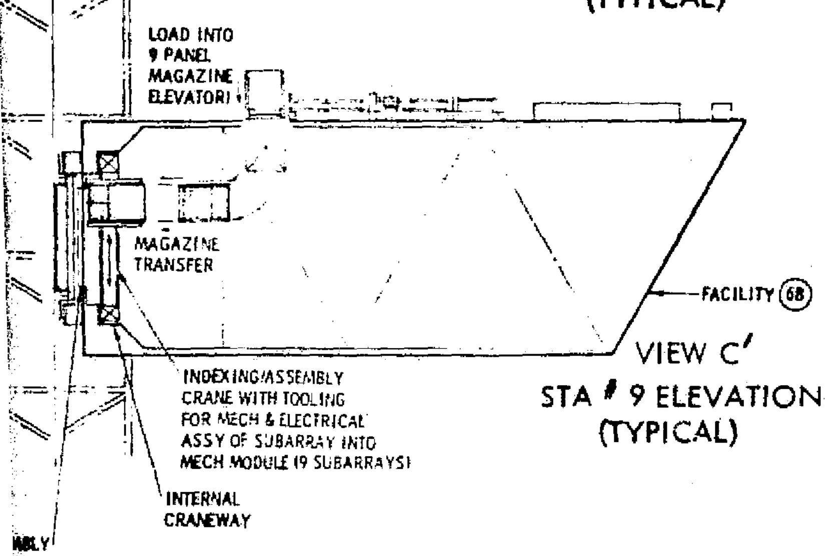


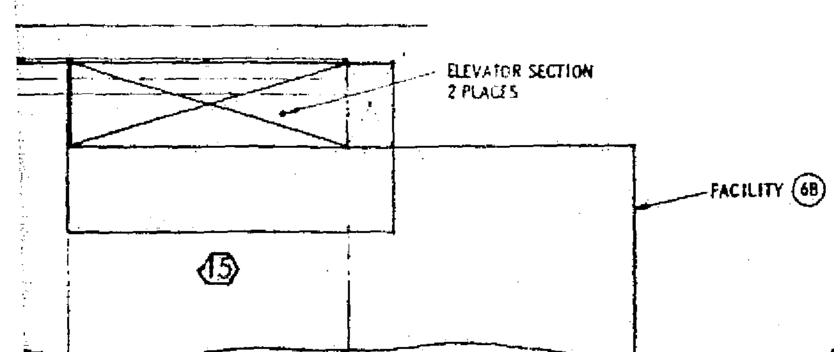
VIEW B'

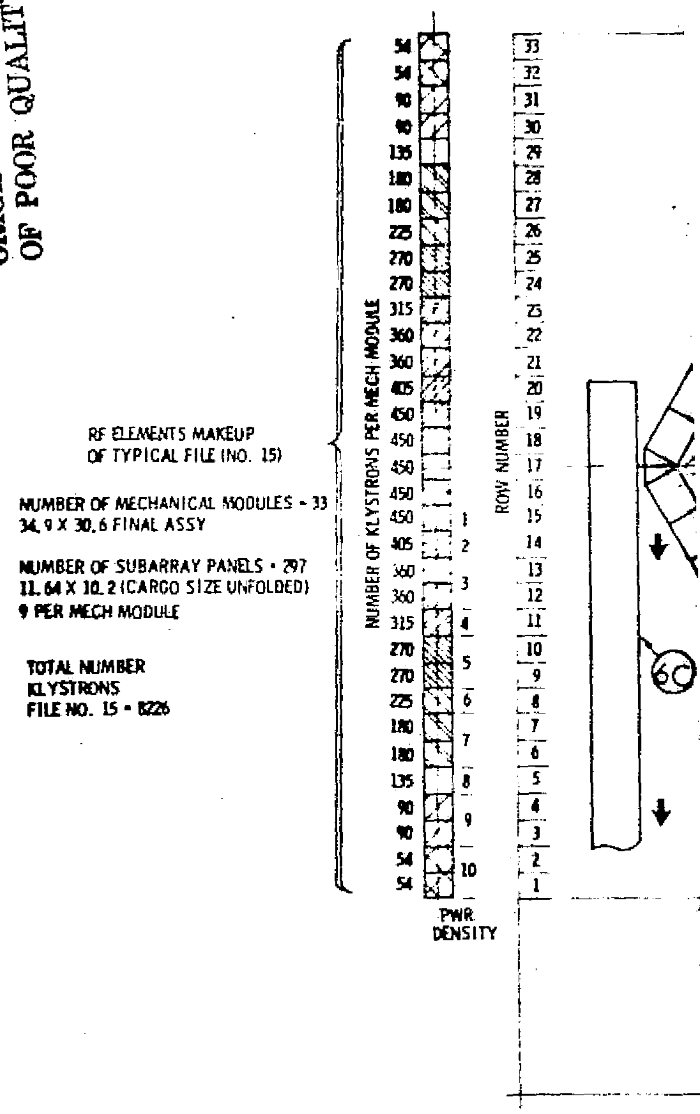
STA # 9 PLAN VIEW (TYPICAL)



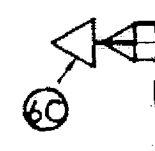
VIEW C'

STA # 9 ELEVATION (TYPICAL)





MW ANTENNA IN RF
ELEMENTS INST. POSITI



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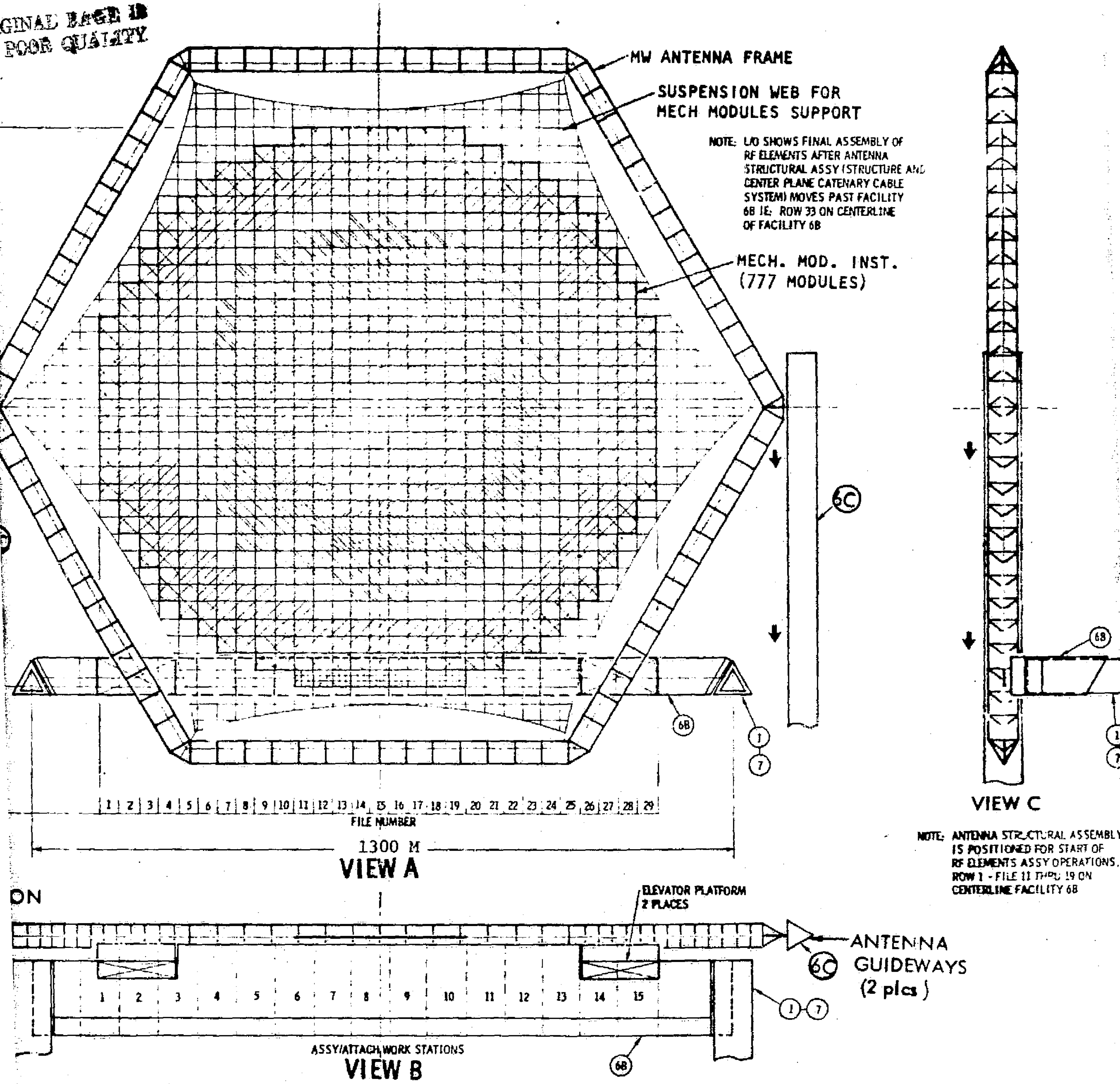

TOP (PLAN) VIEW OF RF ELEMENTS ASS'Y & INST FACILITY

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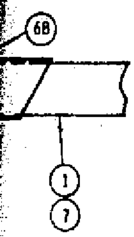
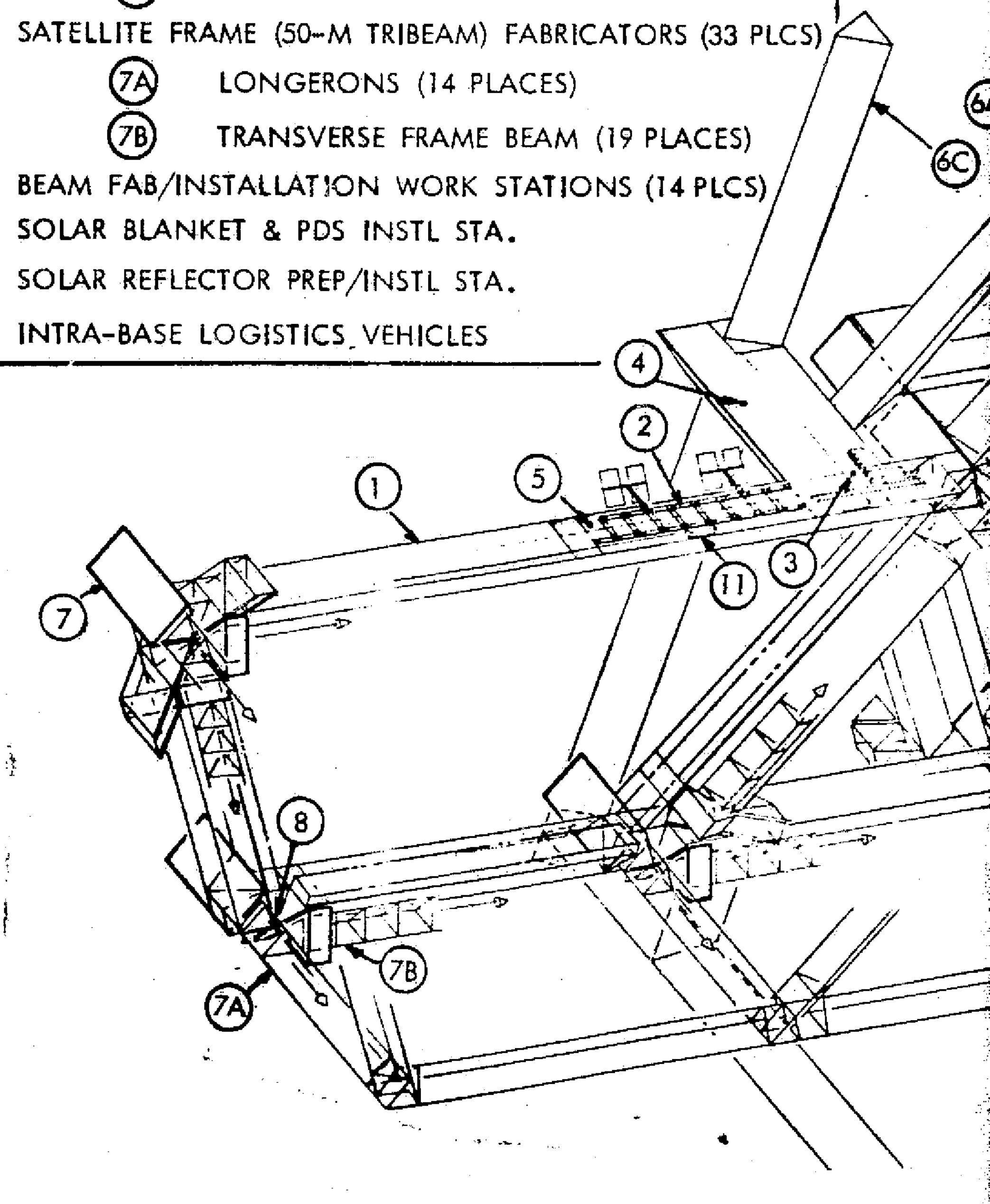


ASSEMBLY
RT OF
RATIONS.
ON

1. CONSTRUCTION FIXTURE
2. BASE SUPPORT FACILITIES & EQUIP (2 EA)
   - 340-CREW MEMBER SUPPORT
   - BASE SUBSYSTEM, MAINTENANCE SHOPS
   - BASE MGMT, COMM., CONTROL, LOGISTICS
3. WAREHOUSE
4. EOTV DOCKING/CARGO RECEIVING
5. POTV DOCKING
6. MW ANTENNA ASSEMBLY FACILITIES
   - 6A FRAME FABRICATION FIXTURE
   - 6B RF ELEMENTS ASSY & INSTL FACILITY
   - 6C FRAME TRANSLATION GUIDEWAY
   - 6D FRAME (50-M TRIBEAM) FABRICATORS
7. SATELLITE FRAME (50-M TRIBEAM) FABRICATORS (33 PLCS)
   - 7A LONGERONS (14 PLACES)
   - 7B TRANSVERSE FRAME BEAM (19 PLACES)
8. BEAM FAB/INSTALLATION WORK STATIONS (14 PLCS)
9. SOLAR BLANKET & PDS INSTL STA.
10. SOLAR REFLECTOR PREP/INSTL STA.
11. INTRA-BASE LOGISTICS VEHICLES



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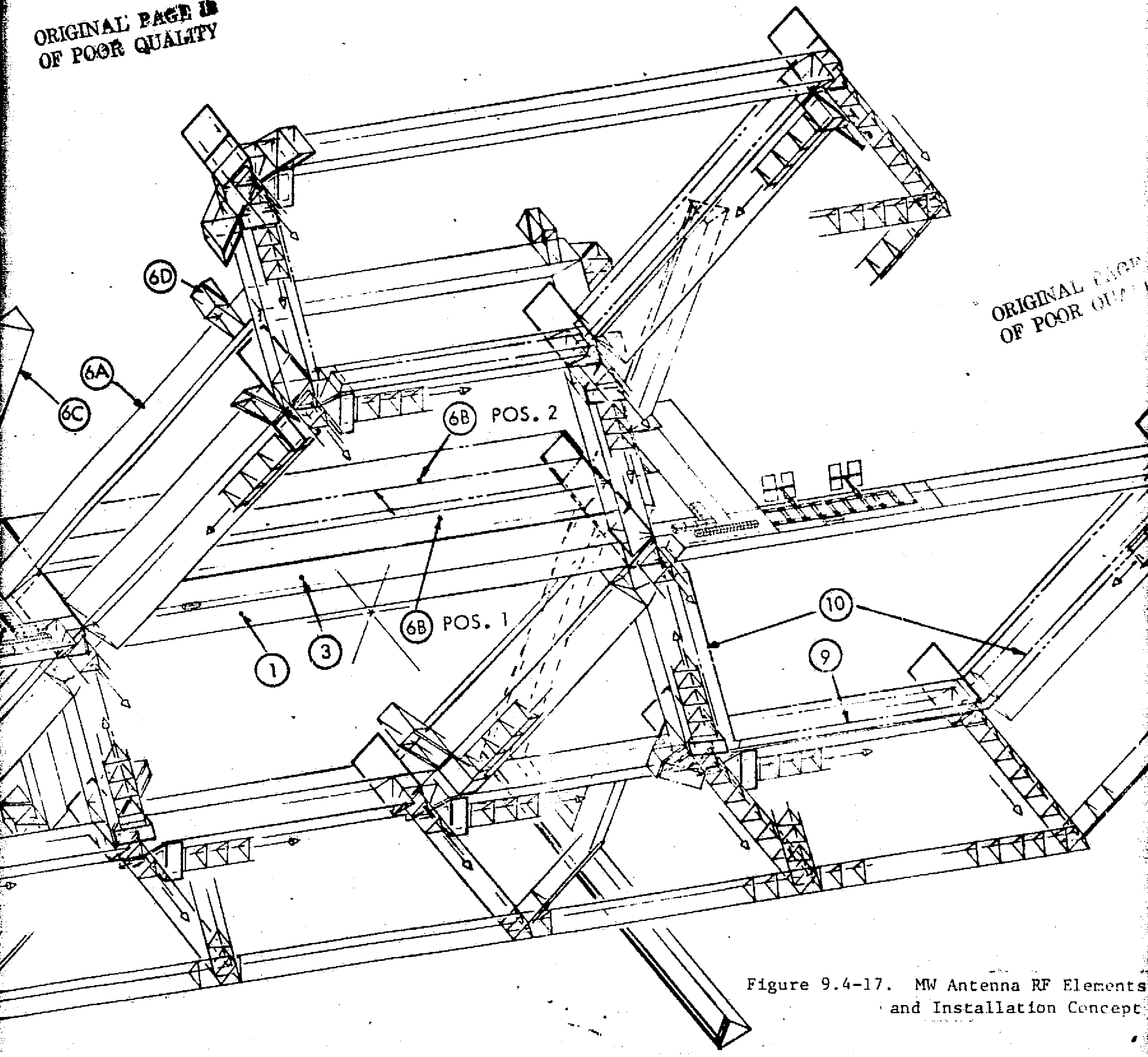

Figure 9.4-17. MW Antenna RF Elements
and Installation Concept

SATELLITE CONSTRUCTION BASE (SCB)
(REFERENCE)

Rockwell International
Space Division

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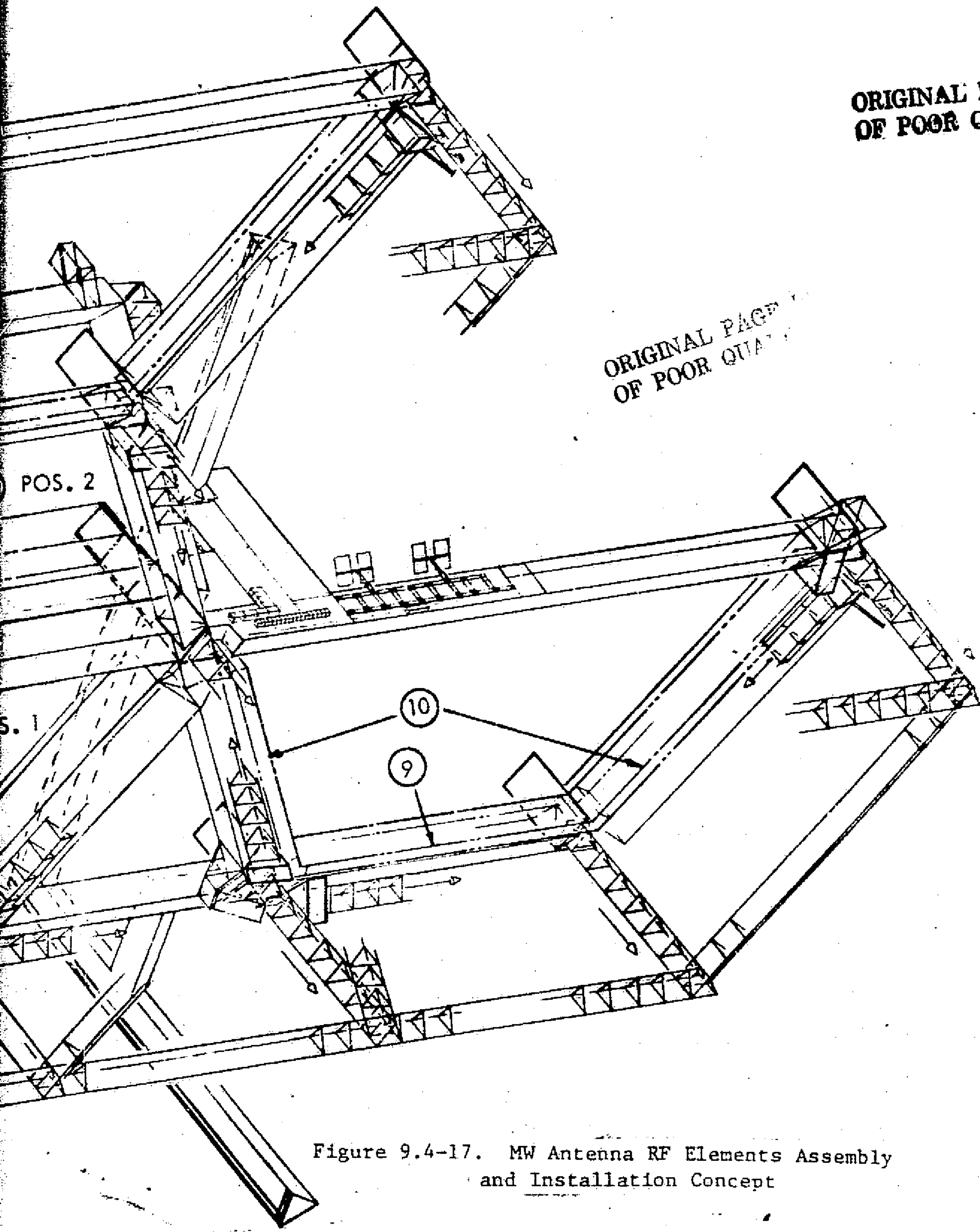

Figure 9.4-17. MW Antenna RF Elements Assembly and Installation Concept

modules from their assembly jig and transfer them straight across to their installation location on the web, the modules remaining parallel to the web at all times. The module transfer units also accomplish the module-to-web-attachment, the electrical hookup and the module-checkout-after installation operations.

A key element of the transfer/installation operations is the minimal distance through which the completed module is moved. The rf facility remains in position 1 (Satellite Construction Base Concept perspective) until the antenna frame translates upward past the facility (moving into row 1 installation position). The rf facility then translates outward to within 10-m of the antenna web (position 2) where it remains until all modules are installed. At that time it moves back into position 1 which allows clearance for the top antenna frame as the completed antenna begins its translation toward the position for installation onto the rotary joint and subsequent checkout (Panels B and C, Figure 9.4-14A).

Conceptually all operations described are automated with crew supervision and remote manual capability as backup.

Rotary Joint Construction

The MW antenna rotary joint provides the means for maintaining the MW antenna rectenna-pointing as the solar collector portion of the satellite rotates through 360 degrees each day to maintain solar orientation. The completed installation is shown in Figure 9.4-18 (the antenna is included for reference). It is located at the center of the satellite between the two solar converter wings. It consists of the antenna mounting trunnions (for azimuth pointing) supported by structural masts which, in turn, are attached to two parallel rotating slip rings 500 m apart. Each of the two rotating slip rings is concentric to a parallel, non-rotating, inner ring. The inner rings are attached to the six longerons which form the hex-section structural core which runs the full length of the satellite. The longerons are stabilized by hex-shaped cross frames at the two sections where the rings are located.

The concentric sets of structural rings are configures as shown in the lower right of Figure 9.4-19. The inner ring is stationary, while the outer ring to which the antenna is secured rotates as required for the desired antenna orientation. Brush blocks and drive assemblies are located between the inner and outer rings on rails installed for that purpose. These assemblies provide for power transmission across the moving surfaces and also the drive mechanism for rotation of the outer ring.

The fabrication sequence is indicated in Figure 9.4-19. A free flying version of the tribeam fabricator is used to construct the ring structure in situ. The inner ring structure is fabricated first. The tribeam fabricator progresses around the hex-section satellite center structure fabricating the 50 m tribeam slip ring structure at the correct radius. Initially, enough of the ring is fabricated to allow fastening to a corner of the hexagonal frame; after it is secured, the tribeam fabricator progresses around the radius to successive corners of the frame, where the beam is secured in a similar manner. The final segment of the ring is fabricated separately and translated radially

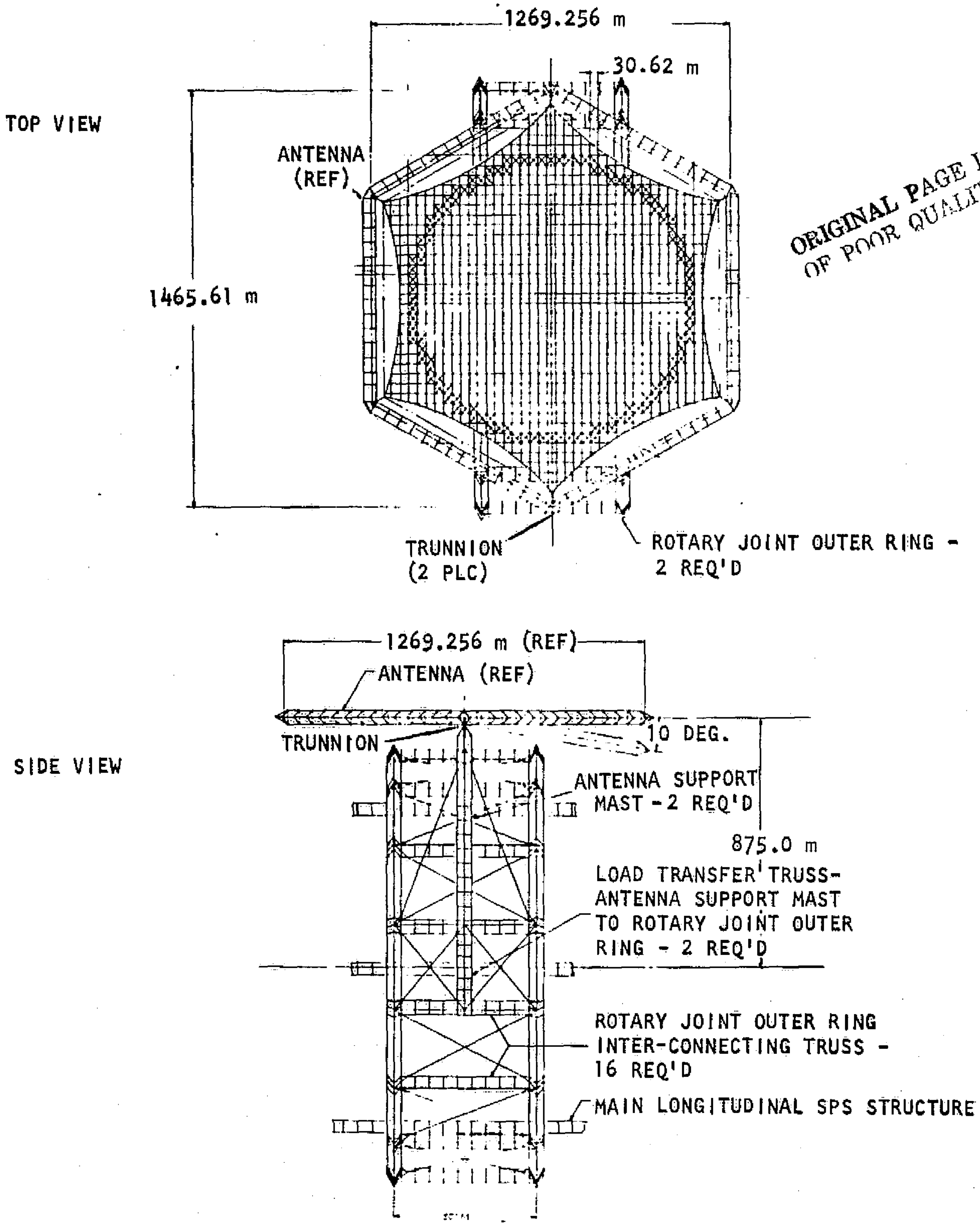

Figure 9.4-18. Rotary Joint

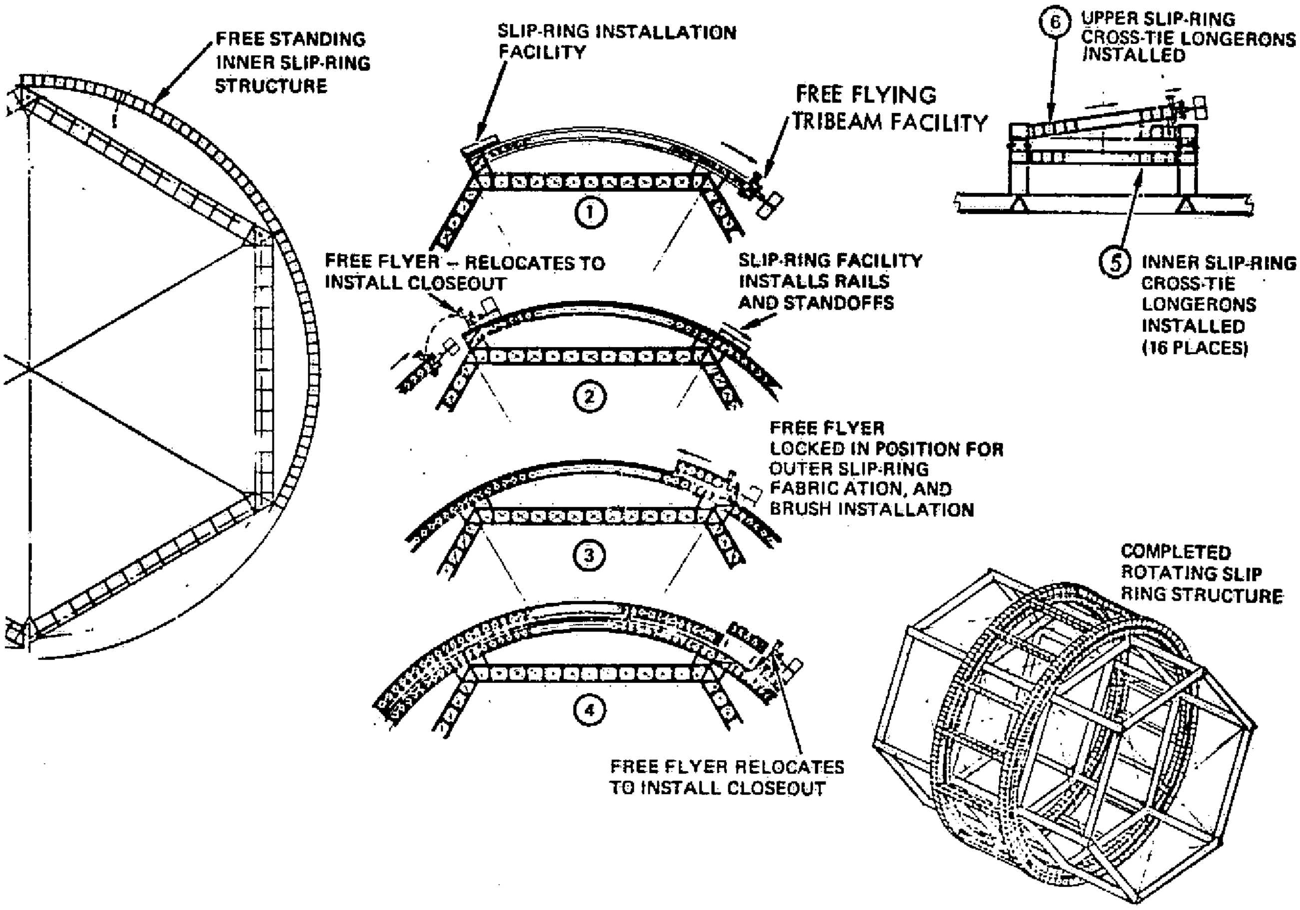

Figure 9.4-19. Slip Ring Fabrication

into place to closeout the ring as indicated. The two inner rings are then connected by 50 m tribeams and cross braced with tension-ties at 16 places around this circumference to provide longitudinal stabilization.

As the inner ring structure progresses, the slip ring tracks are installed in prefabricated segments by a separate facility which operates from the rails after installation of the first segments. This same facility installs the brush boxes and drive mechanisms for the outer ring structure. The tribeam fabricator used for the inner rings is then mounted to the rails where it produces the outer ring structure, in situ, as it is guided around the circumference by the rails.

After the two outer rings have been fabricated and the brush installation is completed, longitudinal beams and tension ties are installed between the circumferences for longitudinal stability, as for the inner rings. The two antenna support masts are then fabricated and secured to the outer ring assembly and the trunnion joints to which the antenna is secured are installed. The antenna is installed as described previously.

9.5 PROPELLANT PRODUCTION AND STORAGE

9.5.1 INTRODUCTION

At its peak, the SPS program will require the construction of four SPS vehicles per year. This production rate will necessitate the placement of material and personnel into low-earth orbit, (Figure 9.5-1). In order to meet this high mass-to-orbit rate, the earth-launch-vehicles (ELVs) will consume even greater quantities of propellant (LH_2 and LO_2). In fact, peak daily ELV propellant consumption is anticipated to be 3500 metric tons of liquid hydrogen and 21,000 metric tons of liquid oxygen. These large amounts of hydrogen and oxygen are approximately 20% and 60% of the present daily U.S. production rate, respectively, and it is important to assess the nations ability to meet these demands.

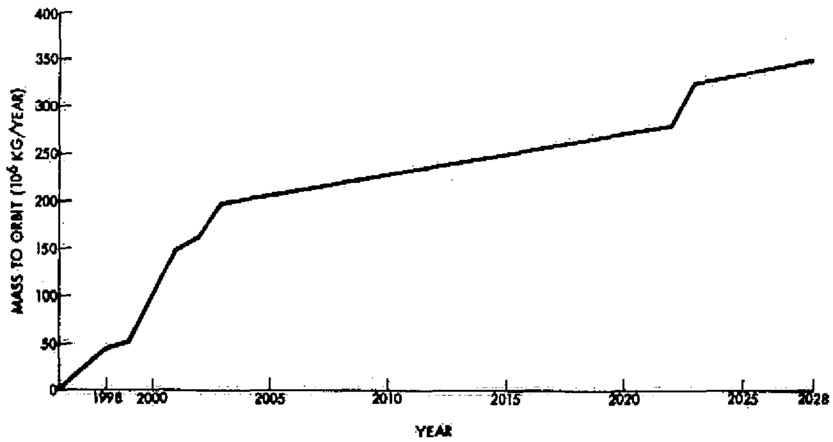

Figure 9.5-1. Mass Delivered To GEO

This study was performed to analyze various techniques available for liquid hydrogen and liquid oxygen production. Figure 9.5-2 summarizes the scope of the analysis.

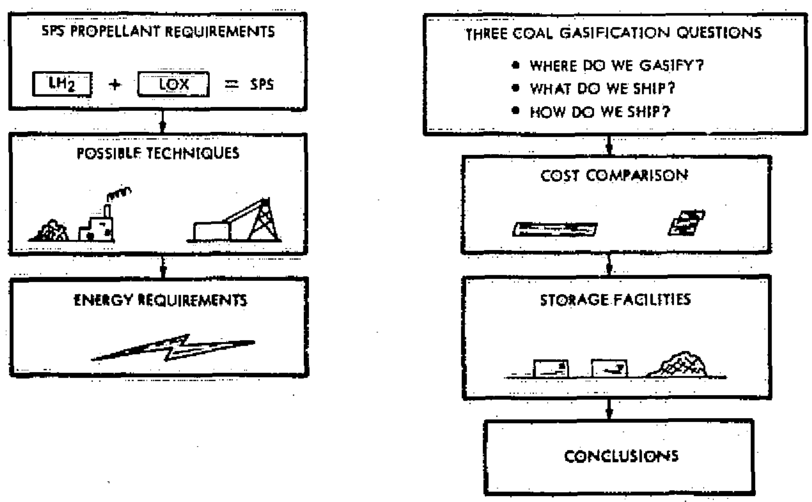

Figure 9.5-2. Scope of Analysis

The ground rules for this study were as follows:

- Launch from Cap Kennedy
- Solar electric OTV with argon as propellant
- Mass-to-orbit as listed in Figure 9.5-1
- Packing factor of 15%

The total mass-to-orbit as a function of SPS production rate can be translated into ELV propellant required as a function of year; and these data are presented in Figure 9.5-3.

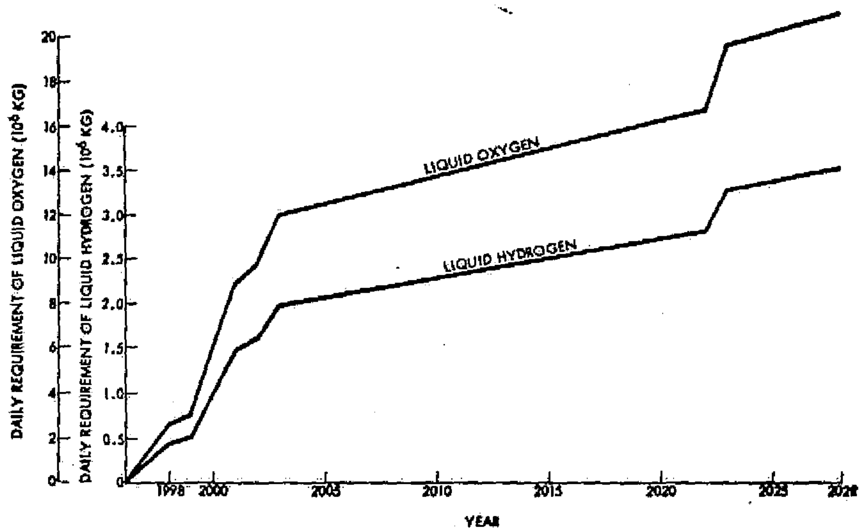

Figure 9.5-3. HLLV Propellant Requirements

There are a number of techniques which can be utilized in order to produce these large amounts of propellant. Currently, liquid oxygen is produced by liquification of air as well as by electrolysis of water. Both these techniques are viable sources of LO_2 for the SPS program.

There are several techniques which may be used in the production of liquid hydrogen. The most feasible alternatives are the production of hydrogen from natural gas, from coal gasification, by the electrolysis of water, by thermochemical processes, and from photosynthetic processes.

Natural gas as a source of hydrogen was considered not to be a viable source. Natural gas is expensive and will be more expensive in the future. It is unreasonable to allow SPS hydrogen production to be dependent upon a natural resource that will be very scarce at the time when the SPS program will require peak hydrogen production.

Thermochemical and photosynthetic processes are awaiting development and there are no assurances that either of these techniques will be able to provide the necessary hydrogen.

The only two techniques which will be capable of providing the required SPS program hydrogen are coal gasification and the electrolysis of water, and these two techniques were analyzed in-depth in this study.

Electrolysis

Figure 9.5-4 presents a block diagram of a typical electrolysis process. The electrical energy for electrolysis can be supplied by a variety of sources; here, it is provided by a nuclear power plant. Desalinized ocean water is split into oxygen and hydrogen, then liquefied and stored. This process has the advantage that for every pound of hydrogen produced, eight pounds of oxygen are produced simultaneously; more than enough to serve as oxidizer for the ELV (mixture ratio 6:1). Thus, both propellants are produced in a single operation and at the same production facility.

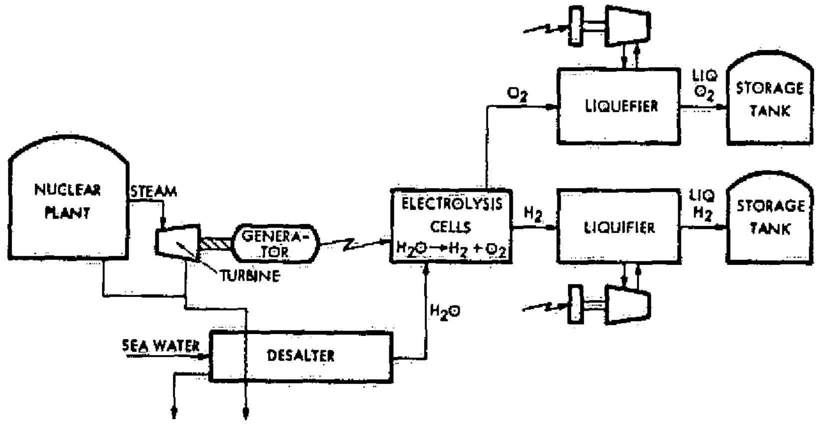

Figure 9.5-4. Typical Electrolysis Process

The power required to produce LH_2 and LO_2 by electrolysis of water is indicated in Figure 9.5-5 (data is for the General Electric solid polymer electrolytic cell). Most of the power is consumed in the splitting of water.

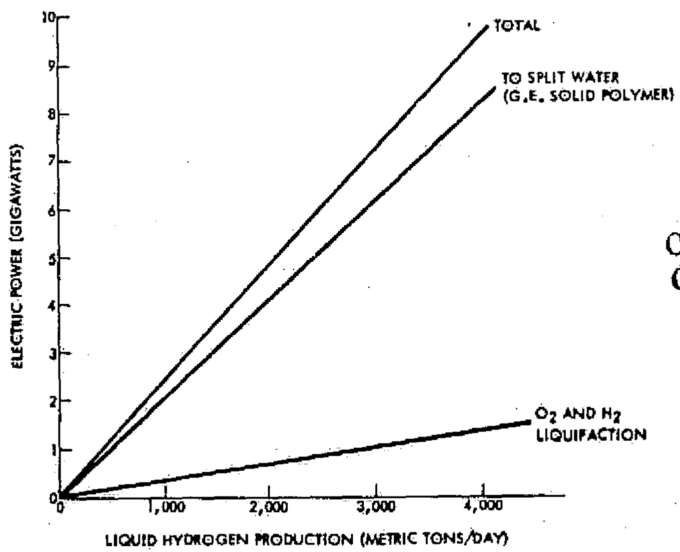

Figure 9.5-5. Electrical Power Required By Electrolysis

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At a hydrogen production rate of 3500 metric tons per day, nearly 8 GW of power are required. This means that, near the end of the SPS production phase, the energy equivalent of nearly two SPS's will be necessary.

Although the power requirements of electrolysis may seem high, the ease of operating such a plant make it an attractive alternative. The plant can be located along the east coast of Florida, thus eliminating logistical problems, and desalinization of ocean water can be accomplished for only a fraction of a percent of the total energy required for electrolysis.

Coal Gasification

Coal gasification, on the other hand, is a much more complicated operation. The schematic presented in Figure 9.5-6 depicts a typical coal gasification process. Pulverized coal is vaporized in the presence of steam and oxygen to release hydrogen. After purification, the gaseous hydrogen is liquefied and stored.

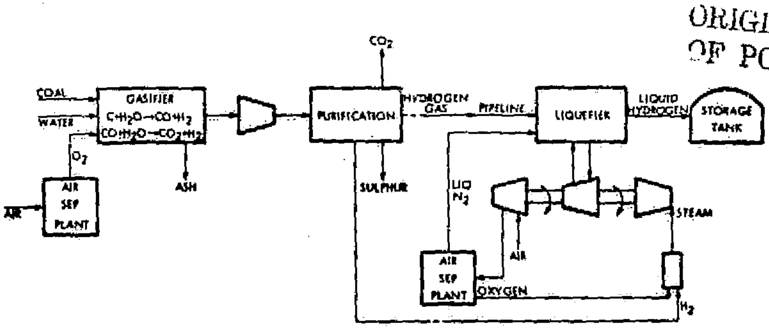

Figure 9.5-6. Typical Coal Gasification Process

This technique, unlike the electrolysis of water, produces only hydrogen and the oxygen necessary for the ELV propellant must be produced by the liquefaction of air.

Coal gasification also produces significant percentages of carbon, carbon dioxide, and other pollutants. Every kilogram of hydrogen produced requires 6.4 kg of coal, 5.3 kg of water, and 6.9 kg of oxygen, and liberates 0.6 kg of carbon and ash. The quantity of oxygen necessary to liberate hydrogen from coal is nearly equivalent to that needed as oxidizer for the ELV flights, and the total coal consumed throughout the SPS program will be approximately 30% of all the coal mined in the U.S. in 1970.

Figure 9.5-7 presents the power necessary for coal gasification along with that needed to produce ELV liquid oxygen. At the peak SPS production rate, the electric power required to produce ELV propellant is approximately 1.5 GW.

A comparison of the energy requirements of electrolysis and coal gasification is presented in Figure 9.5-8. As can be seen, the power needed by coal gasification is nearly one-fifth that necessary for electrolysis.

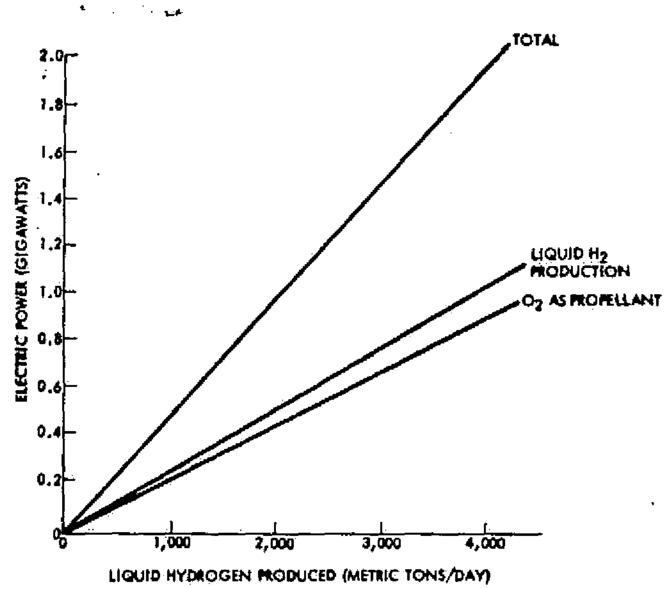

Figure 9.5-7. Power Required For Coal Gasification

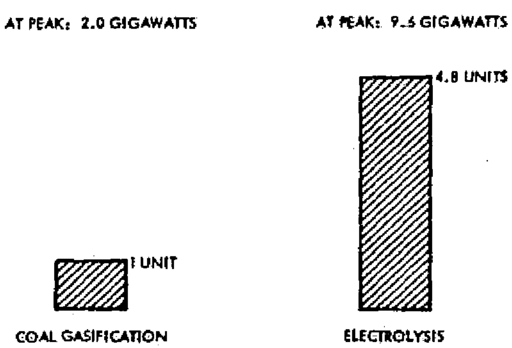

Figure 9.5-8. Process Power Requirements

Logistically, however, coal gasification is more complex than electrolysis, since it would require transporting large amounts of coal or hydrogen over long distances--from the coal mine to Cape Kennedy. It is therefore, important to delve more deeply into the specific logistical alternatives of coal gasification.

The major U.S. coal reserves are located in three geographic areas: the Appalchian region, the Mid-Western region, and the Western region (Figure 9.5-9). The Appalachian coal reserves are essentially committed to eastern energy requirements. This coal is located underground and must be mined using costlier underground mining techniques. The midwestern coal has a high sulfur content and presently cannot meet the pollution standards of most cities, making it nonusable. The western coal is low in sulfur and is essentially undeveloped. It is surface coal and, therefore, relatively inexpensive to mine. Abundance, low sulfur content, and undeveloped nature make the western coal reserves the prime source of coal for SPS hydrogen production.

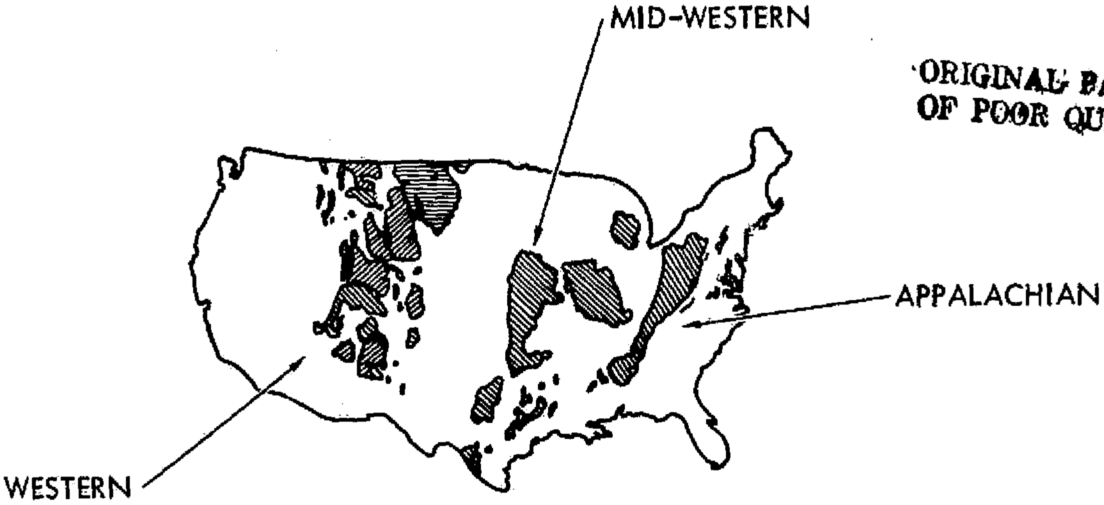

Figure 9.5-9. Geographical Location of U.S. Coal Reserves

However, the location of the western coal reserves necessitates a complicated logistics scenario. There are numerous questions which must be answered in order to develop the most efficient and cost-effective means of handling coal gasification from the mine to the launch site.

The main question is whether coal should be shipped from the mine to the launch site (where it would be gasified), or whether coal should be gasified at the mine and then the hydrogen shipped to the launch facility.

Since coal gasification requires large amounts of water, it may be advantageous to ship the coal to a location with an abundant water supply. It is therefore important to analyze the various alternatives available for transporting coal.

Figure 9.5-10 presents the relative cost of transporting coal by various techniques. Coal slurry is 50% water and 50% coal by weight. The difference between the two coal slurry curves indicates differences in estimated terrain effects and construction costs.

An integral train is a unique concept which does not exist at the present time. It consists of a system of cars which are much larger than conventional train cars, having the capability of quick side-dumping; motors at both ends alleviating the need for turning the cars around for the return trip; and semi-permanently attached cars.

Barging coal is not a feasible alternative since the coal must be barged through the Panama Canal and the long distance involved makes barging too costly.

Figure 9.5-10 indicates that, for the distances considered here, the integral train concept may be the least expensive means by which to transport coal from the mine site to a coal gasification plant (at Cape Kennedy), although coal slurry also may be competitive once further information has been compiled.

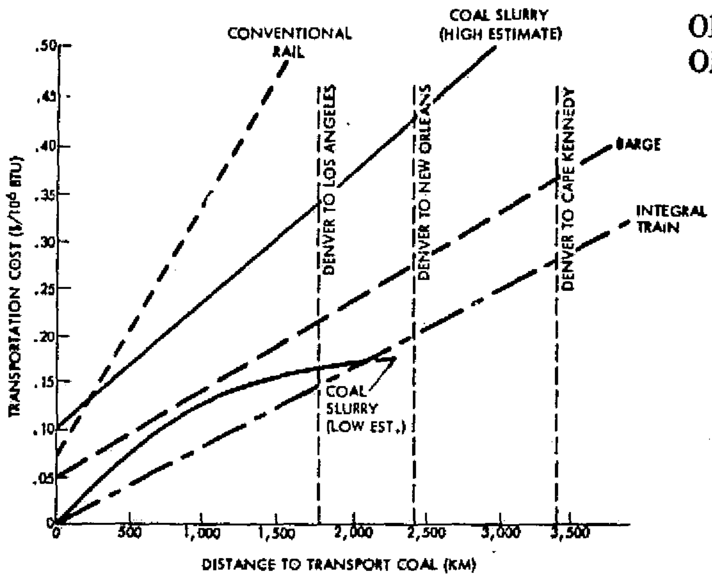

Figure 9.5-10. Cost of Transporting Coal

Another alternative available is to gasify coal at the mine site and ship gaseous hydrogen to the launch site. A comparison of the costs of shipping coal, with those of shipping gaseous hydrogen, is presented in Figure 9.5-11. The two cost curves for shipping gaseous hydrogen result from considering the construction of new pipelines as opposed to using portions of existing natural gas lines.

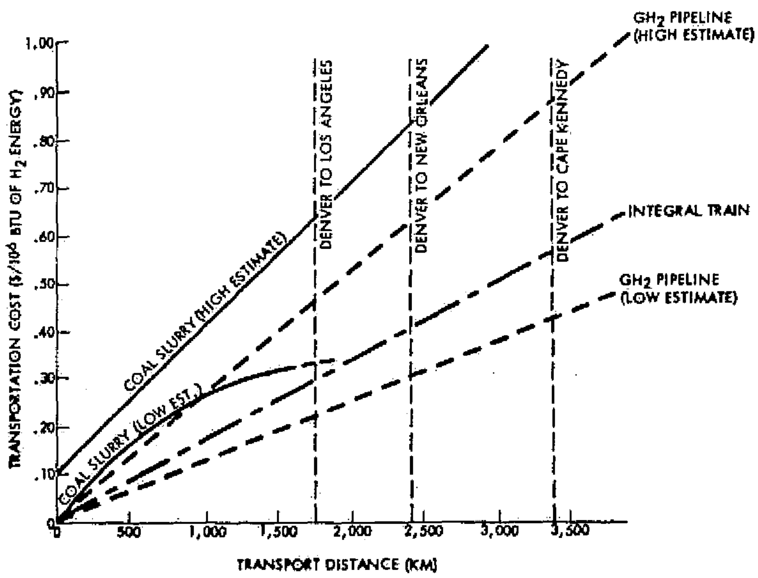

Figure 9.5-11. Relative Transportation Costs

As is evident in the curves, no firm conclusion can be drawn at the present time concerning the most cost-effective technique. Until a more definitive scenario is developed, it is not clear whether coal should be shipped from the mine to the launch site and gasified, or whether hydrogen should be produced at the mine and shipped to the launch facility.

A factor which may influence this choice is the amount of water required by the coal gasification process. The SPS program will require approximately 35,000 acre-feet of water per year for nominal coal gasification production of hydrogen. This is a very small percentage of the total watershed available in the area; although this resource is highly dispersed and not concentrated in rivers and lakes. The watershed is sufficiently large, however, so that by judicious planning the necessary water can be accumulated for coal gassification.

An alternative solution to the water requirement would be to ship water from the Pacific Ocean. Figure 9.5-12 presents the power required to transport water to the western coal region from the west coast. The data indicate that the energy needed is on the order of 0.03 GW, which is a very small percent of the power necessary for coal gasification.

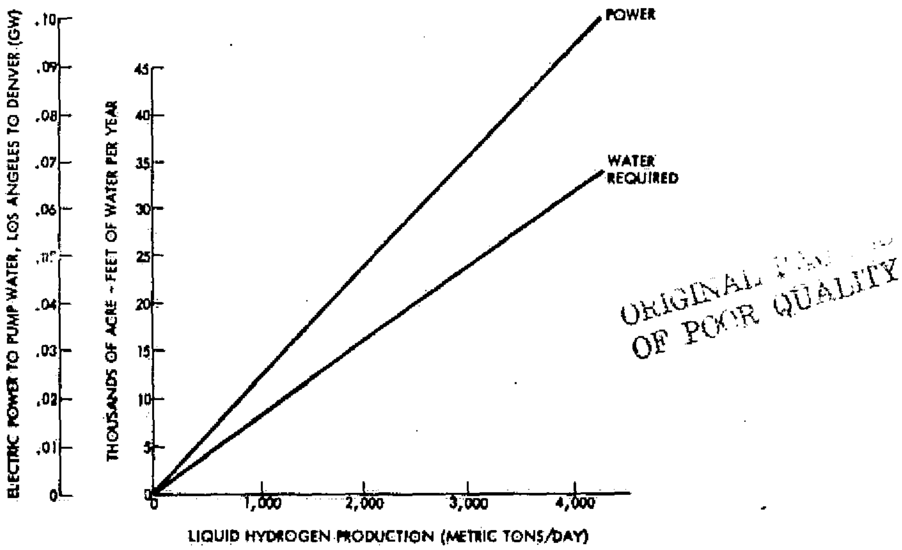

Figure 9.5-12. Water/Power Required For Coal Gasification At Mine

The conclusion, then, is that even if there is insufficient water within the western-region environment, the power necessary to transport it from the west coast is not significant when compared to the total coal gasification power requirement.

Figure 9.5-13 presents a summary of the costs for various alternatives in the production of liquid hydrogen and liquid oxygen. The costs are indicated as the cost of producing one pound of liquid hydrogen and six pounds of liquid oxygen per pound of liquid hydrogen.

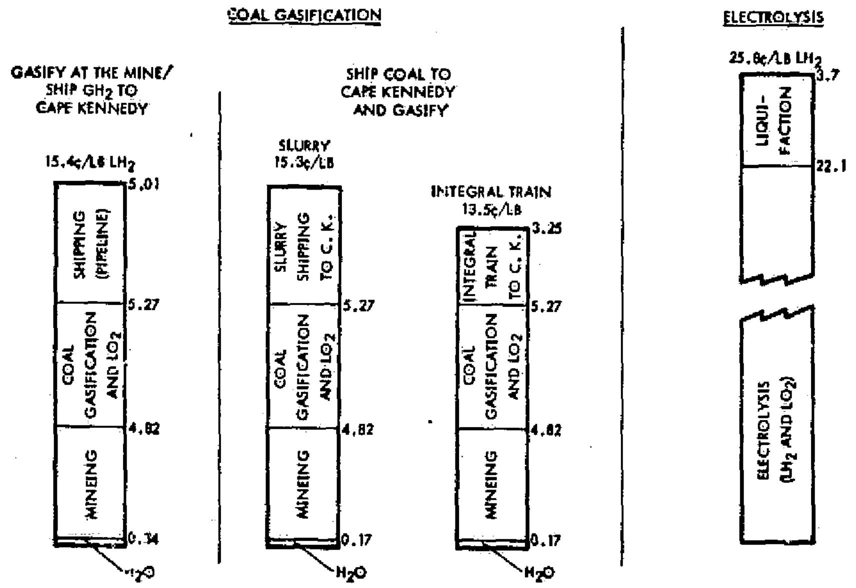

Figure 9.5-13. Cost Summary

Although the integral train seems to be the least expensive alternative for coal gasification at the present time, the uncertainty in the production scenarios precludes making this a final decision.

It is also important to note that, although electrolysis requires five times the power necessary for coal gasification, electrolytic production of propellant is only twice as expensive as coal gasification--after considering the logistical costs of transporting coal or hydrogen from the western coal reserves to Cape Kennedy.

This analysis has not considered environmental factors, ease of operation, maintenance, and other problems unique to a system which transports material 3000 km. It is clear that operational considerations could easily make electrolysis (at the launch site) the most attractive technique.

Regardless of the technique which is selected to manufacture hydrogen and oxygen, a storage facility will be required to absorb the effects of unforeseen circumstances and ensure a smooth HLLV launch schedule. The size of the storage facility will depend on the reliability of the propellant production scenario. Figure 9.5-14 presents liquid hydrogen storage area as a function of storage capacity. These data take into account peripheral dikes and advanced techniques in the construction of liquid-hydrogen storage facilities.

9.5.2 CONCLUSIONS

After analyzing the energy and logistical requirements of the two alternative processes, preliminary production scenarios for each technique can be developed.

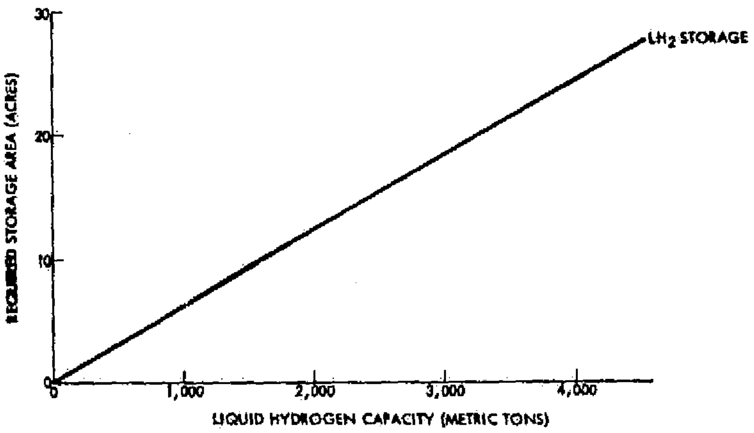

Figure 9.5-14. Liquid Hydrogen Storage Requirements

Coal Gasification Scenario

Figure 9.5-15 presents a block diagram of a possible coal gasification scenario. In this model a coal gasification plant is located at the coal mine and the resulting gaseious hydrogen is piped to Cape Kennedy, where it is liquified and stored until needed. (The liquid oxygen would be produced by liquefication of air and this plant would be located at Cape Kennedy.)

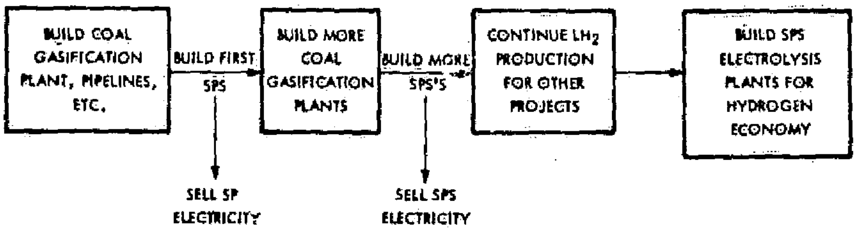

Figure 9.5-15. Possible Coal Gasification Scenario

In the early stages of the SPS program a single gasification plant would be necessary. The electric power from the SPS's would be sold to consumers. As SPS construction continues, more coal gasification plants would be built with all the resulting SPS electrical power sold to consumers.

At the end of the SPS construction phase the coal gasification plants could be used to generate hydrogen for other projects, or perhaps converted to produce other chemicals.

A factor which has yet to be considered in this study, is a concept known as the "hydrogen economy", which is the replacement of hydrocarbon fuels presently supplying transportation, domestic, and industrial energy needs by hydrogen. The transformation to a hydrogen economy will require vast quantities of hydrogen which can only be produced from the electrolysis of large amounts of ocean water.

The coal gasification plants built for the SPS program could play only a minor role in the initiation of the hydrogen economy. A number of SPS's would be needed to provide energy for the electrolytic plants which would be necessary for this future energy source. The coal gasification scenario, therefore, does not lend itself to easy modification and adaptation to possible future pressures.

Electrolysis Scenario

A possible electrolysis scenario would be much more flexible, however, as indicated in Figure 9.5-16. Initially, a nuclear power plant would be used to produce hydrogen and oxygen by electrolysis. The plant would be located at or near Cape Kennedy, and the propellant produced from this first facility would be used to fuel HLLV's for the construction of the first SPS.

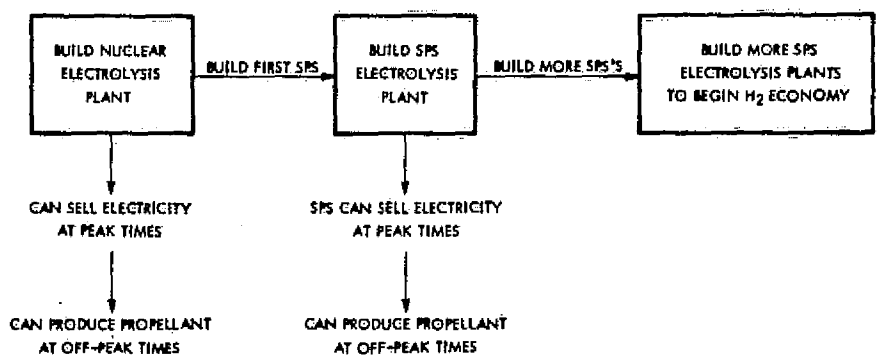

Figure 9.5-16. Possible Electrolysis Scenario

Once an SPS is operational, its power would be devoted to generating electricity for electrolysis--thus producing the propellant for future ELV launches. As the SPS program continues, the electrical energy of the subsequent SPS's could be sold to consumers.

Ultimately, if a hydrogen economy is desired, the electrolysis plants (powered by SPS's) could be used to initiate a smooth transition from the current dependence on hydrocarbon fuels. These electrolytic plants could be located at numerous locations around the U.S.

In the short term, hydrogen production by coal gasification is the least-expensive as well as the least-flexible approach. On the other hand, SPS-powered electrolysis is the cleanest, least logistically complex, and most flexible technique presently available.

9.6 LAUNCH BASE FACILITIES

9.6.1 GENERAL

Launch base facilities must provide for (1) receiving, storage, and processing of materials and propellants; (2) storage of HLLV's sufficient for initial operations; (3) refurbishment and checkout of returning HLLV's; and (4) personnel handling and administration. Figure 9.6-1 depicts the launch site operations and facilities, and their interrelationships, which must be provided for material and personnel processing.

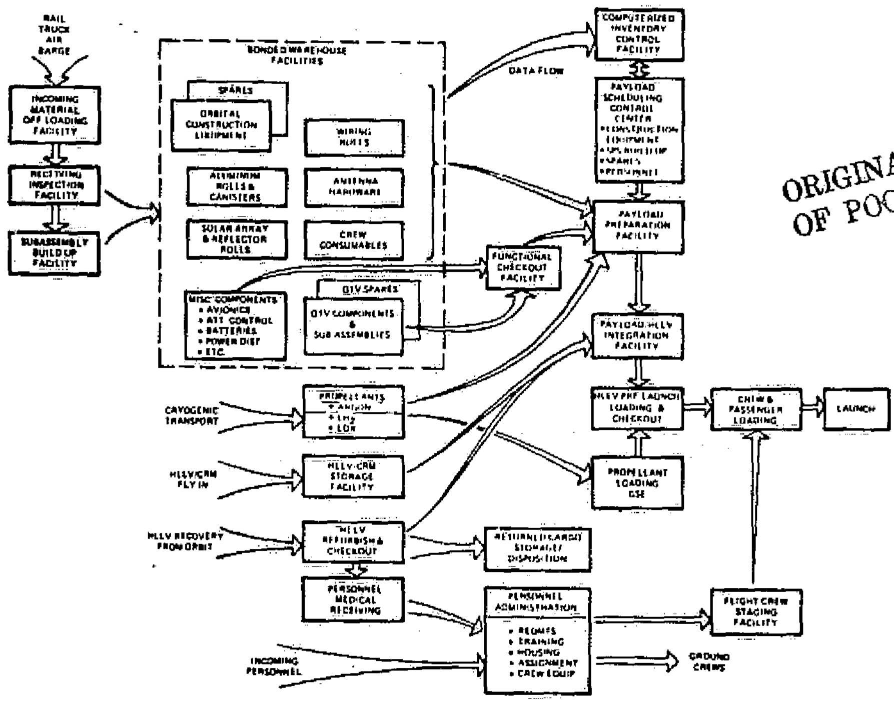

Figure 9.6-1. Launch Site Logistics

Incoming material (rail, air, etc.) is offloaded, subjected to receiving inspection, taken up on the computerized inventory control system, and then stored in the appropriate warehouse facility as shown in the figure. Some of the material will be processed through a subassembly buildup facility prior to storage. This material includes the basic metal stock required to fabricate and assemble the 6993 microwave subarrays required for each SPS. It is estimated that 15,000 m^2 of floorspace will be required for subarray fabrication.

When scheduled by the Payload Scheduling Control Center, materials (construction material, consumables, and spares) are transferred to the payload preparation facility for packaging and arranging into payload units. These integrated payload units are secured on 6×30-m pallets to facilitate orbital handling, transfer, and installation into the 6×6×30-m HLLV cargo bay. Electronic modules and other selected components will be functionally tested prior to packaging. The packaged payloads are then transported to and loaded on the ELV prior to propellant loading and final HLLV checkout. Personnel comprising part of a payload will enter their crew module in the payload bay shortly before launch.

Since HLLV flight requirements are substantial, even in the first year of the program (approximately 550), a fleet buildup will be required prior to initiation of the orbital phase of the program. Hangar facilities, which will be utilized for ELV refurbishment, also can be used (if desired) to house HLLV's.

Personnel administration and logistics, to process and assign/reassign a continual and increasing personnel flow, are vital elements in the overall base function. Incoming personnel must be trained and assigned to either space, flight, or ground crews. Training will be conducted at various locations; much of the training for launch site operations will be conducted at the launch site. Personnel returning from 90 days in orbit must undergo medical processing and then be reassigned to ground activities for a period (TBD) before returning to orbit. Personnel attrition (e.g., resignations, sickness, etc.) may be significant relative to maintaining the required overall number of personnel. The continuous growth in number of both base personnel and space crews throughout the 30-year program precipitates the requirement for extensive facilities for medical, training, and administration.

Finally, provision must be made for processing and disposing of large amounts of packaging materials and failed/damaged hardware which will be returned from orbit by the HLLV's.

9.6.2 STORAGE REQUIREMENTS

Bonded warehousing must be provided for satellite construction material and spares and for COTV construction material. The following assumptions were used in estimating area requirements.

1. Space must be provided initially to accommodate construction material for one satellite and one COTV.

2. Warehouse space must be augmented to support the increasing maintenance mass flow requirements as the number of operational satellites increases.

3. COTV spares can be accommodated in the same area utilized for COTV construction material with no conflict.

4. Beam-machine cassettes will be stored on end; other rolls (solar cell blankets and reflectors) will be stored lengthwise—single tiers in all cases.

5. Microwave waveguide subarrays will be stored on end.

6. A storage density of 1000 kg/m^3, stored to a depth of 2 m was assumed for the remainder of the mass.

7. A warehouse sizing factor of 25 percent above basic storage requirements was utilized to provide for accessibility.

Table 9.6-1 summarizes the storage requirements for the mass of one satellite and one COTV. The COTV storage requirements were obtained by factoring

Table 9.6-1. Storage Requirements Summary

ITEM	QTY	DIMENSIONS (METERS)	UNIT STORAGE AREA(M^2)	TOTAL STORAGE AREA(M^2)	TOTAL REQ'T. WITH 25% FACTOR (M^2)
Cassettes	1188	2(OD)X2.4	4	4752	5227
S/A Blankets	1632	0.6(OD)X25	15	24480	30600
Reflectors	144	1.2(OD)X25	30	4320	5400
MW Panels	6993	11X4.7X0.523(ΔV)	5.753	44254 (with spacing)	55318
Remainder of SPS Mass	16.2×10^6 Kg	1 X 1 X 1 (Store 2M Deep)	1 M^2/ (2000 Kg)	8097	10121
					106666
COTV Mass	17.5×10^6 Kg	(Factor to SPS Requirements Less MW Panels)			27522
				TOTAL	134188

satellite storage requirements by the ratio of COTV mass/satellite mas (excluding microwave panels, since the COTV does not have a microwave antenna). Similarly, the storage required for satellite maintenance material was calculated as shown in Table 9.6-2, predicated on having approximately 20 percent of the annual material spares requirements on hand at any one time. The combined requirements are shown in Figure 9.6-2, illustrating the buildup generated by an increasing number of satellite maintenance sets.

Table 9.6-2. Storage Requirements, SPS Maintenance

YEAR	SPS MAINT. SETS/YEAR	SETS IN STORAGE	STORAGE REQUIREMENTS, M^2 (FACTOR TO SPS REQ'T)
1	0	0	0
2	2	1	1464
5	10	3	4392
10	30	8	11712
15	50	13	19032
20	70	18	26352
25	90	23	33672
30	120	24	35136

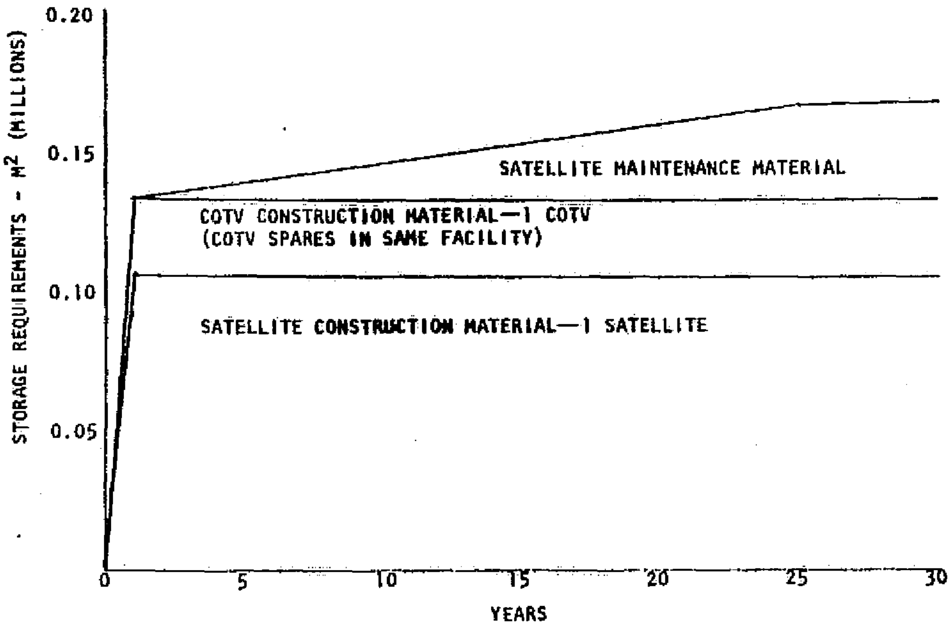

Figure 9.6-2. Launch Site Storage Requirements

Solar array blankets constitute an item for additional investigation. The current package size of 0.6 m, OD, makes no provision for a protective layer. While the need for such a layer has not been definitely established, several materials were considered. One such material, composed of a special foam with a density of 64.07 kg/m^3, can be produced in thickness of 1/32 inch. The addition of this material as a liner for a standard blanket (750×25 m) results in an increase in outside diameter to 1.06 m, and an increase in roll weight to 8091 kg. Current roll weight is 7135 kg with an outside diameter of 0.6 m.

The propellant storage facilities must provide for cryogenic storage of HLLV propellants and for argon which will be shipped to low earth orbit for COTV utilization. Liquid-hydrogen storage area requirements are defined in Section 9.5. The area requirements are based on storage in multi-walled spherical tanks and provides for protective dikes. Ultimately, selected storage capacity will be a function of daily requirements (which will be in excess of 5×10^6 kg/day).

9.6.3 TRANSPORTATION REQUIREMENTS

An analysis of incoming rail traffic necessary to support construction and maintenance of space-based program elements was conducted. The analysis assumed total shipment by rail—although selective use of air, truck, or barge is a viable alternative. Based on Santa Fe railcar specifications, 26.36-m (86.5-ft) boxcars and 26.82-m (88-ft) flatcars with a payload capability of 68,027 kg (150,000 lb) and 136,054 kg (300,000 lb), respectively, were selected in establishing requirements. It is noted that the optimum packing density (100% utilization) for the selected boxcars is approximately 240 kg/m^3, which is exceeded by essentially all of the material allocated to boxcars. Therefore, unused volume usually will exist.

Table 9.6-3 summarizes the rail requirements for one satellite and one COTV. HLLV propellant transportation requirements are included in Section 9.5.

Table 9.6-3. Railroad Transportation Requirements

ITEM	QTY	Dimensions (M)	Item Wt (Kg)	Type Car	No/ Car	Wt/Car Kg	No Cars
Cassettes	1188	2 (OD) X 2.4	2500	Box	11	27500	108
S/A Blanket	1632	0.6(OD) X 25	7136	Flat	6	47400	272
Reflector	144	1.2(OD) X 25	12780	Flat	6	76680	24
Mass (SAT)	19.3×10^6 Kg	1000 Kg/M^3	-	Box	-	68027	340
							744
Mass (COTV)	19.3×10^6 Kg	1000 Kg/M^3		Box		68027	282
							1026

Table 9.6-4 lists yearly requirements for both satellite and COTV maintenance material. Most of the COTV maintenance material consists of cryogenic argon totaling 2.475×10^6 kg per COTV set (10 COTV's). A standard cryogenic tank car with a load capacity of approximately 54,000 kg was specified for argon transportation.

Table 9.6-4 Railroad Transportation, Satellite/OTV Maintenance Material

YEARS	SAT. M. SETS	CARS	OTV M-SETS	CARS	TOTAL CARS
1	0	0	1	47	47
2	2	30	1	47	77
3	4	60	3	141	201
4	7	105	4	188	293
5	10	150	4	188	338
10	30	450	4	235	685
15	50	750	4	235	985
20	70	1050	5	235	1285
25	90	1350	6	282	1632
30	120	1800	7	329	2129

Figure 9.6-3 shows total annual railcar requirements by year required to support the program. The mass required to construct the GEO construction base and LEO support base is not included, but will be defined in the follow-on study. In determining EOTV construction mass, it was assumed that one COTV set would be required for the first two years and would build up to a total of 4-1/2 sets by the end of the program to handle both satellite construction and maintenance mass flow requirements.

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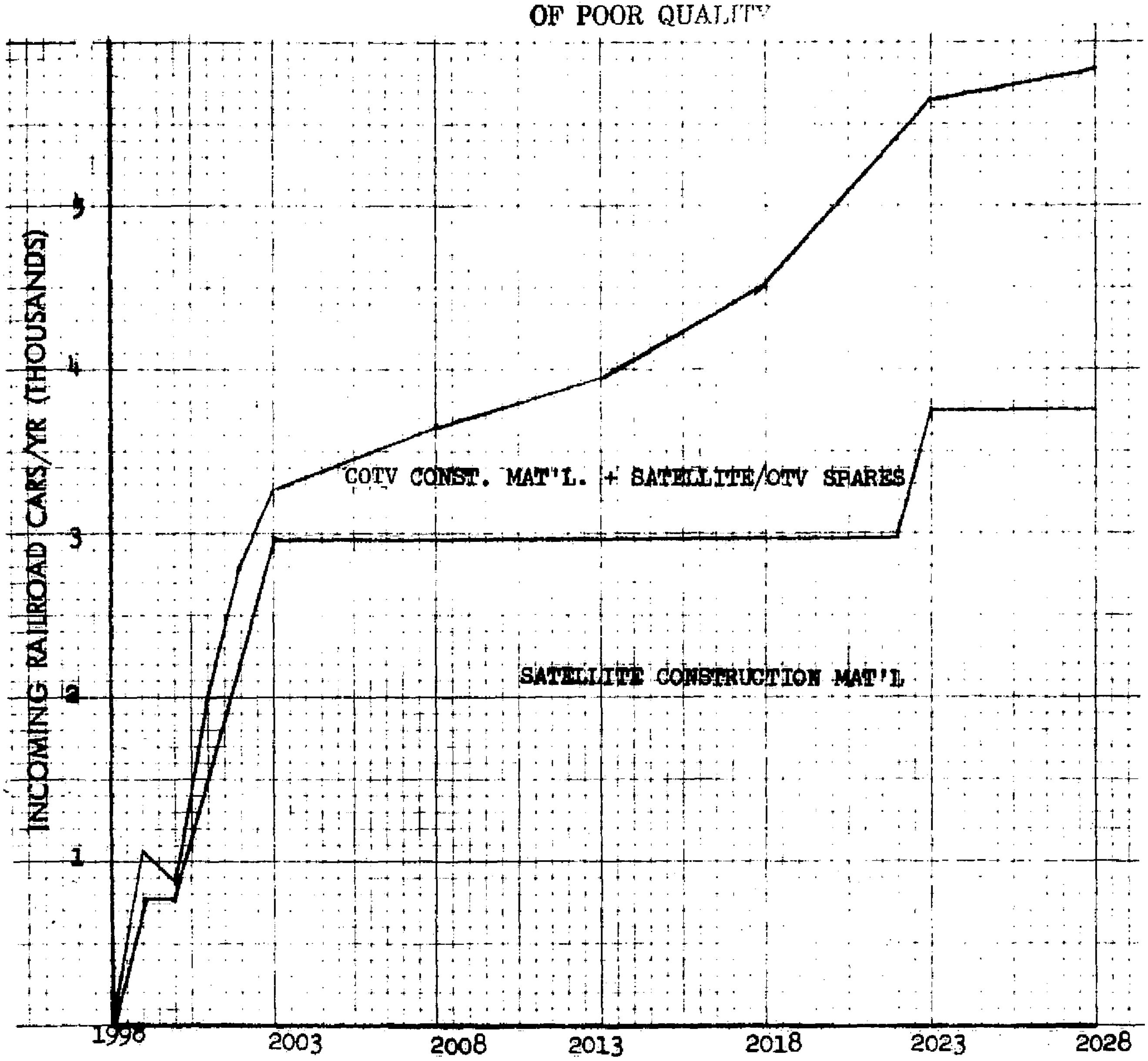


Figure 9.6-3. Annual Railcar Requirements

9.7 PAYLOAD PACKAGING

Payload composition and delivery schedules are influenced by need data in orbit of specific items to support the satellite construction sequence, and by the widely varying number and shapes of items comprising the total construction mass. These factors, coupled with a tacit requirement for most efficient use of the HLLV payload capability (6×6×30 m, 91,000 kg), dictate payload mixes.

Packaging configurations for major satellite construction components, and the quantities required for construction of each satellite, are identified in Figure 9.7-1. These are designed for compatibility with the construction equipment as well as with handling and HLLV cargo bay constraints.

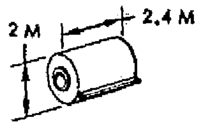
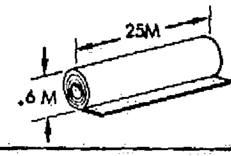
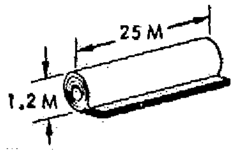
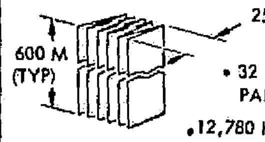
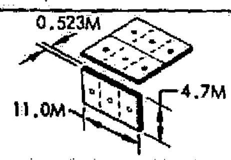
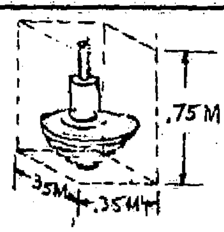
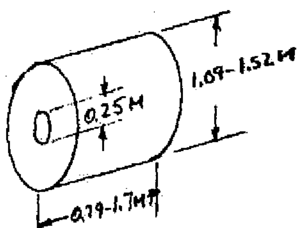
SPS ELEMENT	PACKAGING	PACKAGE DIMENSIONS	NO. REQUIRED	NOTES
STRUCTURES	CASSETTES OF ALUMINUM TAPES		1188	6 DIFFERENT TAPE LENGTHS 2500 KG AVE MASS
SOLAR BLANKETS	ROLLS		1632	750 M LENGTH/ROLL 7136 KG/ROLL
REFLECTORS	ROLLS OF FABRIC-HINGED ALUMINIZED KAPTON SHEET		144	 • 32 "HINGED" PANELS •12,780 KG/ROLL
MW ANTENNA WAVEGUIDE PANELS	SUB ARRAYS		6993	• ALL SUBARRAYS HAVE SAME OVERALL DIMENSIONS • 10 DIFFERENT POWER MODULE SIZES - QUANTITY VARIES WITH SIZE • SUBARRAY MASS (AVE) = 716 KG
ANT.	KLYSTRONS		135864	AV. WT. = 45 kg
PDS	FEEDER ROLLS		162	FIVE FEEDER SIZES WITH AVG WEIGHT = 3375 kg

Figure 9.7-1. Cargo Packaging

Three primary structure cassettes are installed in each beam machine to produce the 2-m triangular beam elements which comprise the basic building block for the 50-m girders. The cassettes contain enough material to complete one half of the satellite structure and must be replaced once subsequent to initial loading in the beam to complete the remainder of the structure. Therefore, sufficient cassettes must be on hand at the beginning of the first wing fabrication to support construction of the entire wing.

Each solar blanket roll is 750 m long—the length required for one bay length. As discussed in Section 9.4.3, a minimum of 68 rolls must be installed in the blanket layer before construction can be initiated. End and side attachment materials and hardware are packaged separately.

Reflector packaging and deployment are described in Section 9.4.3. In addition, klystrons, which do not present a packaging problem, are a major payload item. The microwave antenna contains a large number of subarrays that, in turn, are composed of up to 50 power modules. Each power module has a klystron which is installed in GEO after the subarray has been secured to the antenna.

The aluminum cassettes, solar array blankets, and reflector rolls must be scheduled early in the traffic mode, since wing construction commences at approximately the eighth day of the 90 days allocated for fabrication and checkout of each satellite. The waveguide subarrays have different need dates, but their unique characteristics complicate mass flow planning. For example, an average subarray measures 11×4.7×0.523 m and weighs 716 kg. Since a maximum of only 22 of the required 6993 subarrays can be accommodated in any one payload, it is necessary to include some arrays in almost every payload—resulting in a reduction of cargo bay volumetric efficiency.

Delivery requirements to meet the construction schedule (Figure 9.4-1) are defined by the demand schedule of 9.7-2. Payload composition and delivery

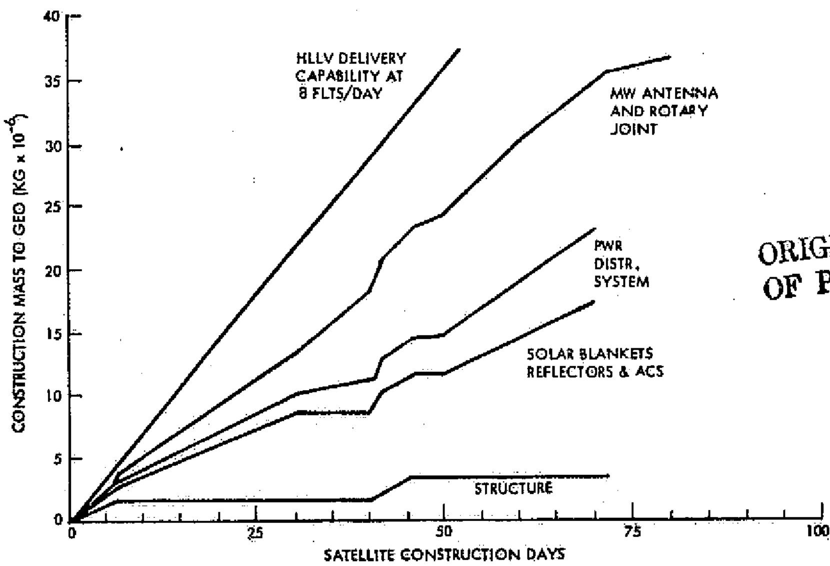


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Figure 9.7-2. Mass Flow Demands for Satellite Construction schedule must support the demands of the individual elements of Figure 9.7-2. The major construction components have been integrated into nine payload configurations which are compatible with the demand schedule and which also make full utilization of the HLLV 91,000-kg cargo capacity. These configurations are shown in Table 9.7-1. The nine payload configurations account for 345 of the 409 HLLV launches required to support construction of one satellite. The remainder of the payloads are comprised of miscellaneous material such as brushes, slip ring segments, etc., and are delivered by 64 additional flights.

Three of the nine payload configurations are shown in Figure 9.7-3. Some unused cargo bay volume remains because optimum packing density for the cargo bay is 84 kg/m^3, which is exceeded by most of the component packages. The notable exception to the high-density components is the waveguide panels which have an average packaging density of about 24 kg/m^3.

Sequencing of the identified payloads is necessary to support the construction schedule. This sequence, iterated to arrive at a payload ordering which supports both the mass delivery requirements and the construction sequence, was developed and is shown in Table 9.7-2. The configuration numbers refer to the configurations identified in Table 9.7-1.

The satellite mass delivered to LEO must be transferred to COTV's for transit between LEO and GEO. The current COTV configuration requires ten vehicles, each with a capability of about 3.9×10^6 kg, to transport one satellite mass to GEO. Since an COTV roundtrip (LEO-GEO-LEO) is about 162 days, a set of ten vehicles is required to support the 90-day satellite construction schedule. A schedule of eight HLLV flights per day was postulated to support the COTV traffic model. This schedule is within the projected launch rate capability, considering other requirements such as maintenance material and crews, and results in satellite mass delivery in 51 days. This is 21 days ahead of the required completion, thus providing considerable margin for contingencies which could slow delivery rates. Figure 9.7-4 shows the mass flow demands of the major satellite elements plotted against GEO deliveries resulting from the payload sequencing of Table 9.7-2, assuming a departure of one of the ten required COTV's every 5+ days. In all cases, the delivery schedule meets or exceeds the requirements.

Table 9.7-1. Payload Configurations

CARGO CONFIGURATION	CARGO CONTENTS	TOTAL UNITS REQUIRED EACH WING/SPS	NUMBER OF FLIGHTS	UNITS PER FLIGHT	TOTAL THIS CONFIGURATION	UNITS PRIOR FLIGHTS	TOTAL UNITS DELIVERED	UNITS TO BE DELIVERED	UNIT WEIGHT (Kg)	UNIT VOLUME (M^3)	TOTAL THIS FLIGHT WEIGHT (Kg)	TOTAL THIS FLIGHT VOLUME (M^3)	CUMULATIVE FLIGHTS	CUMULATIVE MASS (Kg × 10^6)
1	ST CASSETTES	594/1,188	91	13	1,183	-	1,183	3	2,500	9.6	32,500	125	91	8.28
	SA ROLLS	816/1,632		6	546	-	546	1,086	7,136	9.0	42,816	54		
	MW PANELS	6,993		20	1,820	-	1,820	5,173	716	29.6	14,320	592		
	KLYSTRONS	135,864		30	2,730	-	2,730	133,134	45	0.092	1,350	3		
											90,986	774		
2	ST CASSETTES	594/1,188	1	3	3	1,183	1,186	0	2,500	9.6	7,500	29	92	8.37
	SA ROLLS	816/1,632		8	8	546	554	1,078	7,136	9.0	64,224	72		
	MW PANELS	6,993		20	20	1,820	1,840	5,153	716	29.6	14,320	592		
	KLYSTRONS	135,864		113	113	2,730	2,843	133,021	45	0.092	4,950	11		
	CABLE ROLLS	.743 Kg		16				.673x10^6	438	0.4	7,008	7		
											91,001	711		
3	REFLECTOR ROLLS	72/144	30	3	90	-	90	54	12,780	36.0	38,340	108	122	11.1
	SA ROLLS	816/1,632		4	120	554	674	958	7,136	9.0	28,544	36		
	MW PANELS	6,993		18	540	1,840	2,380	4,613	716	29.6	12,888	533		
	MAIN FEEDER	30/60		2	60	-	60	0	4,620	2.02	9,240	4		
	KLYSTRONS	135,864		44	1,320	2,843	4,163	131,701	45	0.092	1,980	4		
											90,992	685		
4	REFLECTOR ROLLS	72/144	27	2	54	90	144	0	12,780	36.0	25,560	72	149	13.557
	SA ROLLS	816/1,632		6	162	674	836	796	7,136	9.0	42,816	54		
	MW PANELS	6,993		18	486	2,380	2,866	4,127	716	29.6	12,888	533		
	#2 FEEDER	24/48		1	27	-	27	21	3,441	1.62	3,441	2		
	KLYSTRONS	135,864		140	3,780	4,163	7,940	127,921	45	0.092	6,330	13		
											91,005	674		
5	SA ROLLS	816/1,632	21	8	168	836	1,004	628	7,136	9.0	57,088	72	170	15.468
	MW PANELS	6,993		20	420	2,866	3,286	3,707	716	29.6	14,320	592		
	#2 FEEDER	24/48		1	21	27	48	0	3,441	1.62	3,441	2		
	SW GEAR	540/1,080		50	1,050	-	1,050	30	240	1.64	12,000	82		
	KLYSTRONS	135,864		92	1,932	7,943	9,875	125,989	45	0.092	4,140	9		
											90,998	757		
6	SA ROLLS	816/1,632	12	8	96	1,004	1,100	532	7,136	9.0	57,088	72	182	16.56
	MW PANELS	6,993		20	240	3,286	3,526	3,467	716	29.6	14,320	592		
	#3 FEEDERS	18/36		3	36	-	36	0	2,795	1.23	8,385	4		
	SW GEAR	540/1,080		2	24	1,050	1,074	6	240	1.64	480	4		
	KLYSTRONS	135,864		144	1,728	9,875	11,603	124,261	45	0.092	6,480	14		
	#4 FEEDER	6/12		1	12	-	12	0	4,224	1.89	4,224	2		
											90,977	688		
7	SA ROLLS	816/1,632	59	9	531	1,100	1,631	1	7,136	9.0	54,224	81	241	21.931
	MW PANELS	6,993		20	1,180	3,526	4,706	2,287	716	29.6	14,320	592		
	KLYSTRONS	105,864		500	29,500	11,603	41,103	94,761	45	0.092	22,500	46		
											91,044	719		
8	SA ROLLS	816/1,632	1	1	1	1,631	1,632	0	7,136	9.0	7,136	9	242	22.022
	MW PANELS	6,993		22	22	4,706	4,728	2,265	716	29.6	15,752	652		
	KLYSTRONS	135,864		1,000	1,006	41,103	42,103	93,761	45	0.092	45,000	92		
	CABLE ROLLS	.743 Kg		35	35	.007 Kg	.22 Kg	.52x10^6	438	0.3574	15,330	13		
	6 #5 FEEDERS	3/6		6	6	-	-	0	1,795	0.8	7,770	5		
											90,988	771		
9	MW PANELS	6,993	103	22	2,266	4,728	6,994	0	716	29.6	15,752	652	345	31.395
	KLYSTRONS	135,864		910	93,730	42,103	135,833	31	45	0.092	40,950	84		
	1 CABLE ROLL	.743 Kg			103	.22 Kg	.67 Kg	.07 Kg			438	0.4		
	MISC MASS											<344		
											91,000			
10	MISC MASS		64								91,000	<1,080	409	37.2

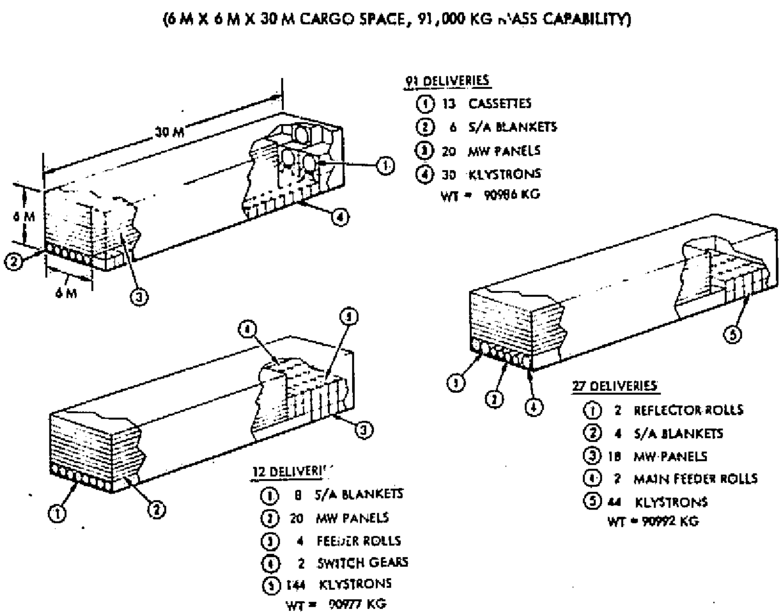

Figure 9.7-3. Representative Integrated HLLV Payloads

Table 9.7-2. HLLV Payload Sequencing

PAYLOAD CONFIG.	NO. OF FLIGHTS	CUM FLTS.	CUM. MASS (Kg X 10^6)
1	40	40	3.64
3	30	70	6.37
2	1	71	6.46
6	6-	77	7.01
10	2	79	7.19
1	25	104	9.47
10	10	114	10.37
5	21	135	12.28
1	26	161	14.65
4	27	188	17.11
6	6	194	17.66
8	1	195	17.75
9	30	225	20.48
10	20	245	22.30
7	59	304	27.67
9	30	334	30.40
10	32	366	33.31
9	43	409	37.2

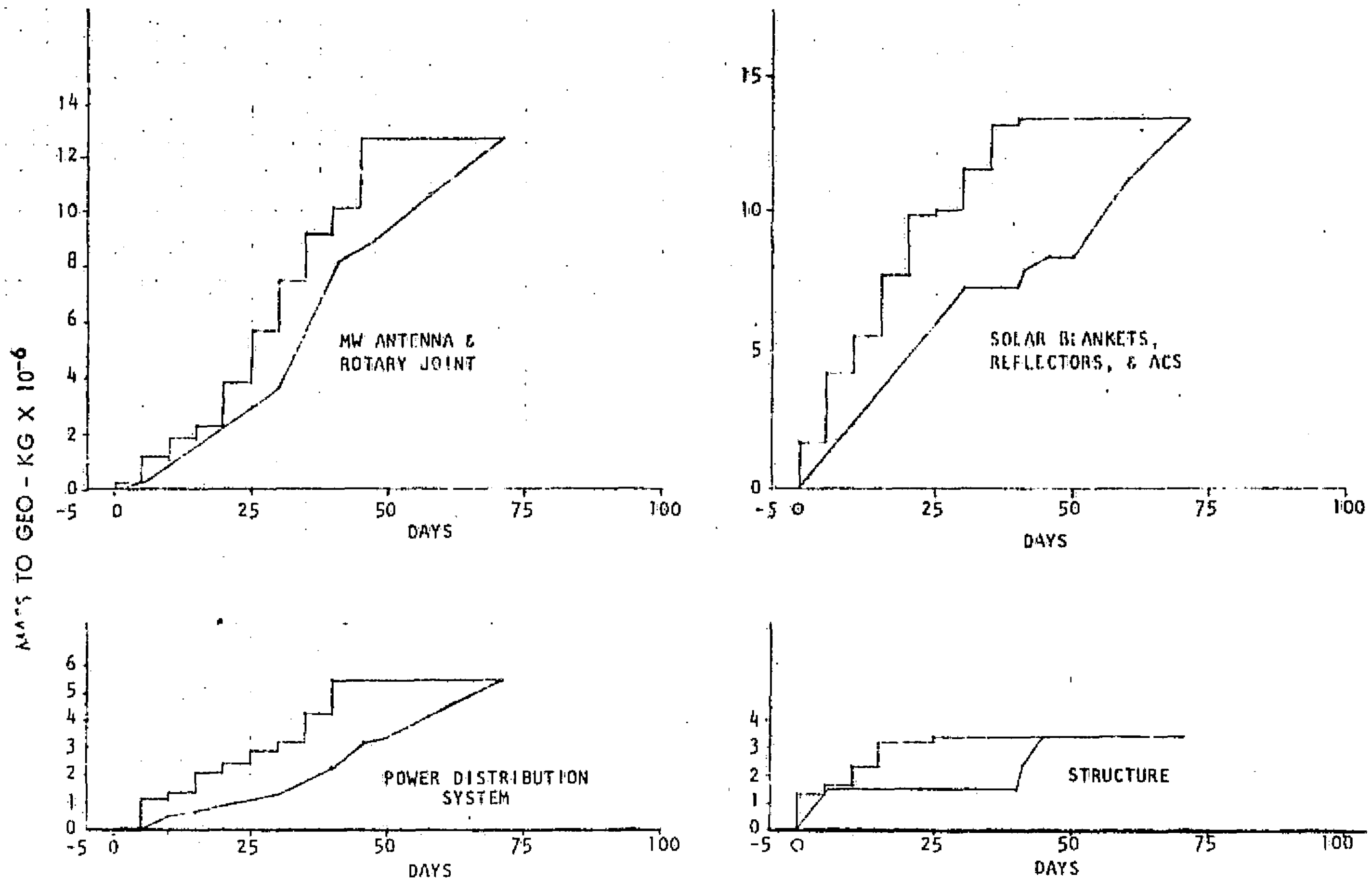

Figure 9.7-4. Mass Flow Demands Vs. Deliveries

9.8 SATELLITE OPERATIONS AND MAINTENANCE

9.8.1 OPERATIONS AND MAINTENANCE SUPPORT REQUIREMENTS

Support requirements consist of personnel and material for maintaining the COTV's, satellites, intra- and inter-facility vehicles, construction equipment, and the orbital bases. The construction sequence of the orbital bases and definition of vehicles required has been identified as a task in the follow-on study. Personnel requirements have been summarized in the preceding section.

The maintenance material required annually for each satellite is summarized in Table 9.8-1 and totals 0.931×10^6 kg. The 10% allowance for packaging increases this to 1.02×10^6 kg. As more satellites are placed in operation, this figure will increase, approaching 122×10^6 kg per year towards the end of 30 years.

Table 9.8-1. Annual Spares Requirements for Each Satellite

SPARES	DATA BASE MASS FOR COMPLETE SATELLITE (KG x 10^{-6})	ESTIMATED SPARES REQUIREMENTS (%)	SPARES MASS (KG x 10^{-6})
KLYSTRONS/WAVEGUIDES	6.52	5.0	0.326
POWER DIST. & CONTROLS	3.77	4.0	0.158
SOLAR BLANKETS	7.76	1.0	0.078
REFLECTORS	1.22	0.1	0.001
ROTARY JOINT	1.82	5.0	0.091
ATTITUDE CONTROL HARDWARE	0.06	5.0	(NEG)
ACS PROPELLANTS	0.06	100.0	0.06
STRUCTURE	2.02	0.1	0.002
SUBTOTAL			0.716
30-PERCENT GROWTH			0.215
TOTAL YEARLY SPARES/SATELLITE			0.931

By comparison, the COTV maintenance mass, consisting of argon tanks, thruster grids, and parts for unscheduled maintenance, is small. The 268 thruster grids required for each of 10 COTV's totals 10,720 kg. The argon and argon tanks for 10 COTV's total approximately 2.5×10^6 kg. This total mass will require about 28 HLLV flights to deliver the material to LEO, where the maintenance activity takes place. The number of COTV maintenance sets required per year will gradually increase, reaching a peak of 7 per year during the last five years of the 30 year program.

9.8.2 OPERATIONS AND MAINTENANCE CREW SIZE

Satellite maintenance activities are anticipated to be essentially a continuous operation. Although the basic concept is to design for a 30 year life, the extremely large number of components (e.g., 135,864 klystrons) results in a high probability of random failures. In addition to component replacement, it is probable that sections of solar blankets must be removed and replaced because of either meteoroid damage or part failures which result in total or partial loss of an arrays' output. Based on probable maintenance activities, a maintenance crew size of 20 has been established. This crew size estimate will be iterated as required to reflect the results of follow-on study analyses.

The maintenance crew must be rotated on a 90-day cycle. The ultimate personnel transportation requirements are considerable. When the number of satellites in operation approaches 120, up to 2400 people must be maintained on station and rotated.

COTV's will undergo maintenance after each LEO-GEO-LEO transit. Scheduled maintenance activities will be accomplished in LEO. These operations consist of replacing thruster grids and replacing the exhausted argon propellant tanks with full tanks. A total of 268 thruster grids and 3 argon tanks per COTV must be replaced. In addition, it is probable that some unscheduled subsystem maintenance must be accomplished. A LEO crew of 30 has been established for this purpose and again, is subject to future iterations. In the event that unscheduled maintenance of a large magnitude is necessary, the basic crew will be augmented by an HLLV personnel flight. The normal maintenance turnaround has been established at 9 days. Since the COTV flights to GEO and return are approximately 5 days apart, no more than 2 COTV's normally will be in the maintenance cycle at any one time.

9.8.3 SATELLITE CONTROL BASE CONCEPT

The permanent satellite operations and maintenance base is installed on each satellite after completion of the center section primary structure or about 50 days after start of satellite construction. The base concept is shown in Figure 9.8-1. The location, near the center of the satellite provides best access to all parts of the satellite and is in close proximity to the MW antenna, which is expected to require a substantial portion of the maintenance effort. The base has facilities for both the crew and for storage of maintenance material, installation equipment and intra-facility vehicles.

The functions of the five modules which comprise the base are identified in the figure. The crew hab module internal configuration and overall dimensions are the same as for the construction base shown in Figure 9.4-4. The crew support module also has the same internal function as for the construction base but occupies only 1/2 of the module, the other 1/2 being an integrated multi-crew member EVA preparation and airlock station.

The maintenance crew, upon arrival, supports installation of the antenna control electronics and satellite checkout, scheduled for from day 50 through day 69.

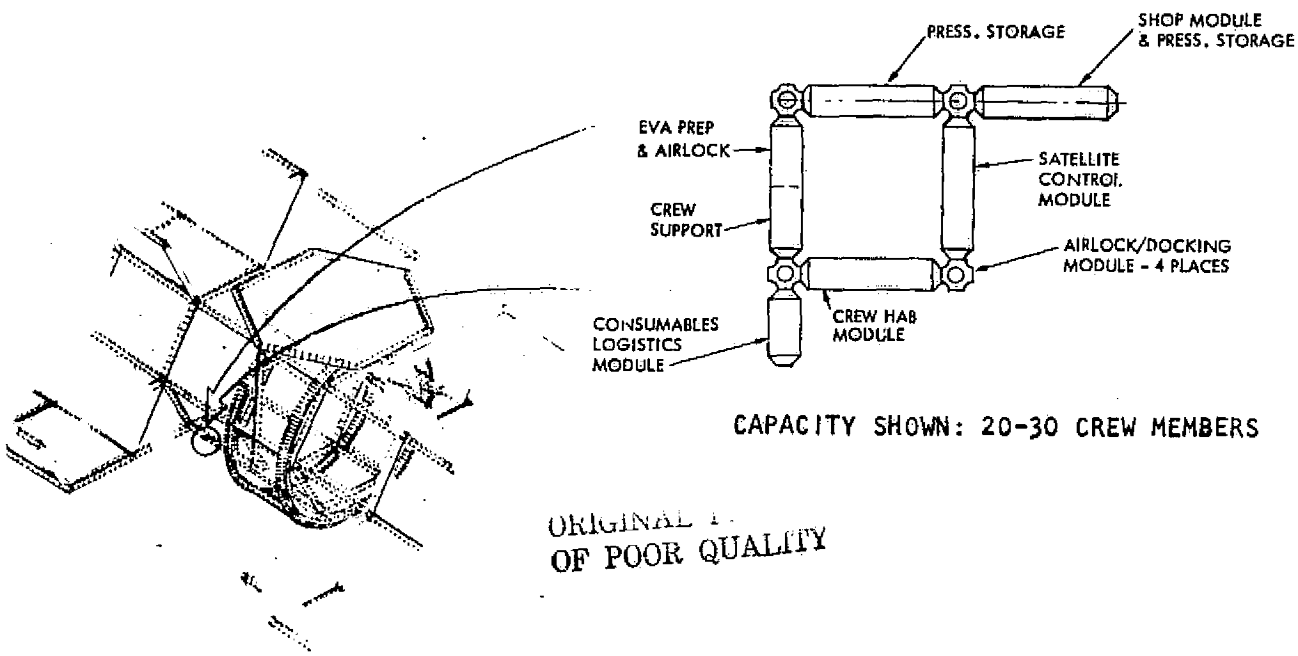

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Figure 9.8-1. GEO Satellite Operations & Maintenance Base

After final checkout and acceptance of the satellite on day 86, the construction crew and equipment are transferred to the next satellite site and the maintenance crew assumes control.

9.9 RECTENNA CONSTRUCTION AND LOGISTICS

9.9.1 RECTENNA CONSTRUCTION

An analysis has been conducted to develop concepts for construction of a 5-gigawatt receiving antenna (rectenna) located at a nominal 34° north latitude site. Based on a 1-kilometer microwave antenna diameter using a 10-step Gaussian beam distribution, the projected intercept area on earth will be a 10- by 13-kilometer ellipse at the 1-milliwatt perimeter of the beam. A typical rectenna site planview would be similar to that depicted in Figure 9.9-1.

Rectenna Panels

The baseline rectenna panel is 12.24 meters wide by 14.69 meters long. A laminated module of the rectenna panel consisting of a stripline pattern of bow-tie dipole antennas etched on a copper faced mylar sheet is shown in Figure 9.9-2. This is a high-dipole-density panel module; three additional module configurations of lesser dipole density are also used. Twenty of the modules shown would make up a single rectenna panel. The other three configurations would require 15, 12 and 10 panel modules to fill the area of the baseline rectenna panel.

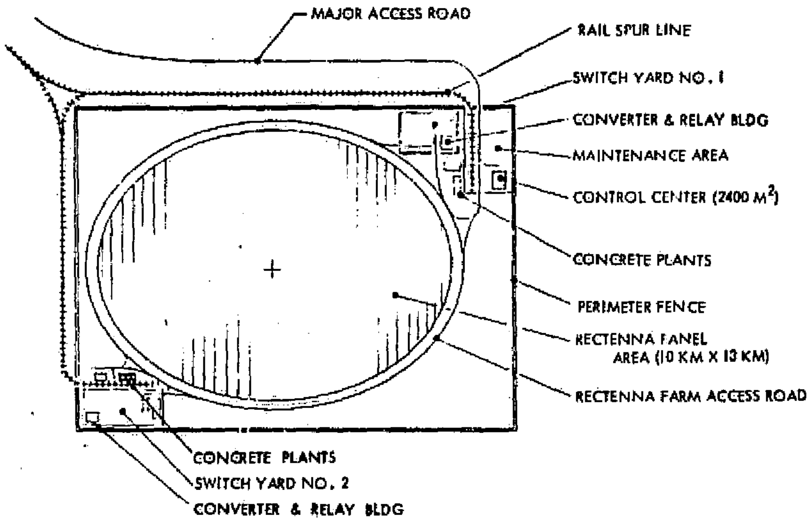

Figure 9.9-1. Rectenna Site Concept

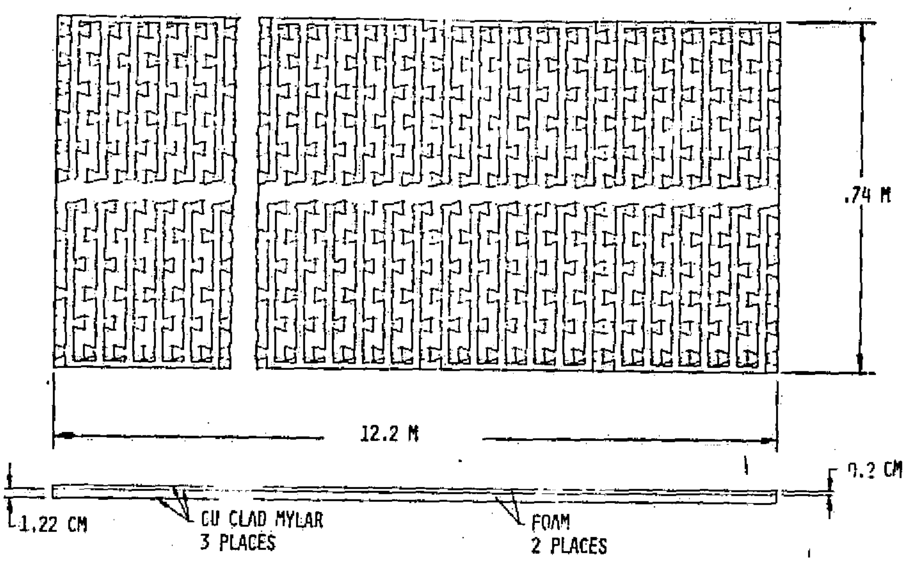

Figure 9.9-2. Rectenna Module
High Density Area

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At the selected latitude of 34°, the required satellite microwave antenna tilt will be approximately 6°; thus, the angle between the horizon and the rectenna panels will be ∿ 40°. Within the intercept area, there will be 436,818 rectenna panels aligned in 814 rows. Since a panel module weighs approximately 2 kg/m^2, then the total mass of rectenna panel modules is 157.085×10^6 kg, or approximately 360 kg/rectenna panel.

Rectenna Panel Support Structure

The laminated foam and copper-clad mylar rectenna panel modules are solid sheets which maximize resultant structural loads due to wind. A special support structure has been designed to react the loads resulting from wind velocities up to 90 mph while holding overall panel deflections to less than 6 centimeters and localized panel deflections to less than 3 centimeters. The concept selected was one which employs thin-sheet (.020 inches) performed hat sections; standard sized (8-inch) I-beams; and 3½-inch diameter, 0.226-inch wall thickness tube braces. The material is galvanized steel. The 24 hat sections are riveted to 4 I-beams which, in turn, are bolted to the tubular braces as shown in Figure 9.9-3. The I-beams and braces support the structure on concrete piers. To allow for adjustments, a screw jack is used at the base of the support braces. The support structure weights are tabulated in Table 9.9-1. With rectenna panel modules, the total weight is approximately 2080 kilographs (4576 lbs).

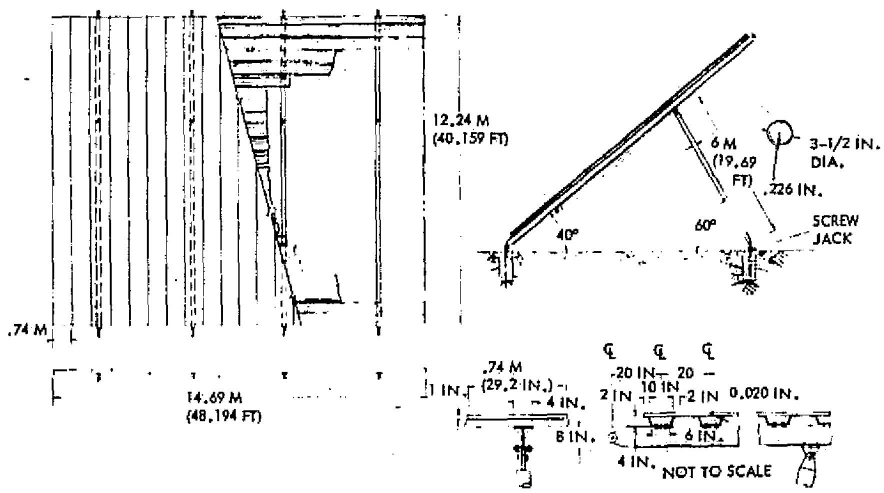
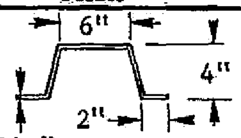
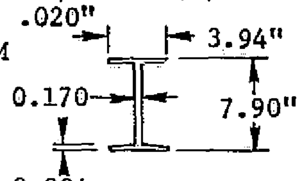
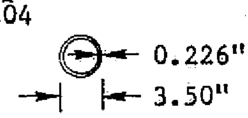


Figure 9.9-3. Rectenna Panel Support Structure

Site Preparation

For rectenna construction, it has been assumed that the site has been boundary surveyed and topographically mapped from aerial photographs, a site plan has been engineered, construction contracts drawn up, and the land has already been acquired or is under option for immediate purchase at the date of aauthority-to-proceed (ATP) with construction. The surveying, mapping and planning is common practice for land purchases of this magnitude; therefore, upon receiving an ATP, the initial contracts can be negotiated and site preparation work can immediately proceed.

Table 9.9-1. Rectenna Panel Support Structure Weights

STRUCTURAL ELEMENT	DIMENSIONAL DATA	NUMBER REQUIRED	WEIGHT (KG)
HAT SECTIONS	14.69M LONG 	24	642.82
I-BEAMS	12.24M LONG 	4	730.17
TUBE BRACES	6.0M LONG 	4	302.06
FITTINGS	-		45.45
		TOTAL WEIGHT	1720.50

Survey teams will move on site and begin correcting the aerial topographical maps and updating the engineering plans. Timber will be clear cut and the land grubbed of stumps and other debris. For relatively flat terrain, work may be initiated along an east-west line in the center of the site in order to minimize the impact of any accruing measurement errors. As the center of the rectenna area is cleared, survey flags can be set and grading operations can begin. Paralleling these operations are the construction of access roads, rail spurs and utility lines; the installation of concrete plants; and the building of maintenance facilities and fencing of construction materials storage yards.

The length of time required for site preparation will vary depending on many factors; e.g., proximity to transportation avenues, general terrain, forested area, and especially the availability of trained laborers. Costs will rise substantially if, in order to compress schedules, heavy equipments and trained operations have to be "imported" in great numbers. In this brief analysis, it has been assumed that a contract could be negotiated which would trade the sale potential of cut timber for the cost of clearing and grubbing the land. Rough manpower and cost estimates indicate that this is a more than equitable assumption given adequate time for these operations. All site preparation work will be a one-shift operation and time-to-completion has been estimated to take approximately 12 months.

The parallel operations and schedules for site preparation are shown in Figure 9.9-4. The manpower estimate footnoted, i.e., 200 man-years, is approximately 94% for site grading. At this state of analysis, it is important to note that in the overview of site preparation and construction, the

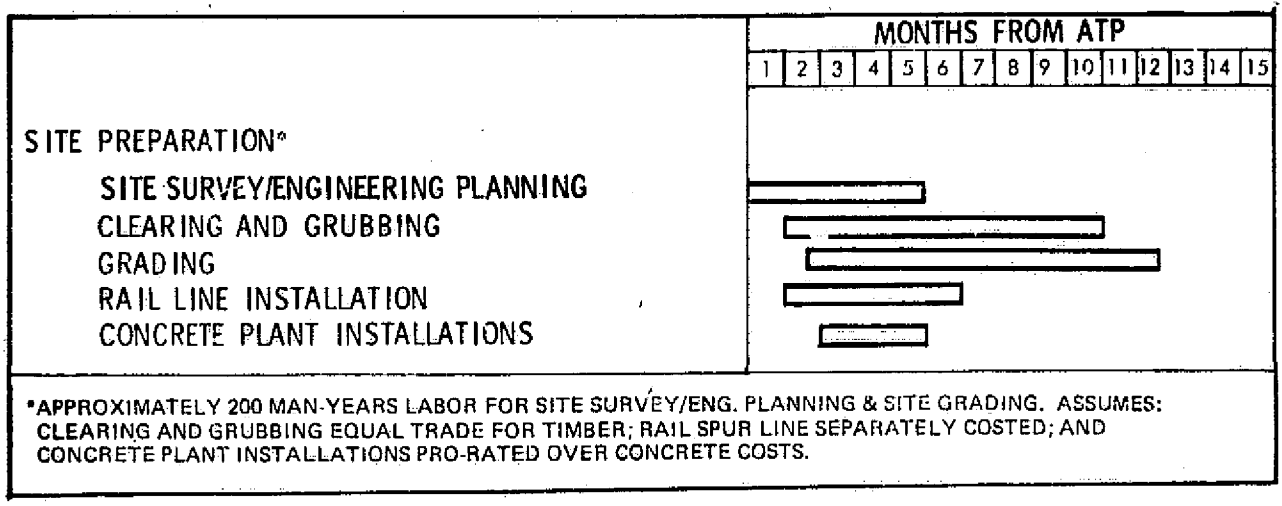


Figure 9.9-4. Rectenna Site Construction Schedule

total capital investment for the rectenna is relatively insensitive to site preparation costs. The dominant concern for site preparation is time-to-complete. The most appropriate way to alleviate that concern in an early study phase would be to select a candidate siting area which has been topographically mapped from aerial photographs and conduct an in-depth analysis of the operations and time requirements to prepare the site for rectenna construction.

Construction Operations

As soon as an area of the rectenna site has been partially cleared, graded, survey lines set and the first concrete plants brought into production, the operations of digging holes for pouring footings to support the 436,818 rectenna panels can commence. It has been estimated that a crew of 2 men can operate and excavate at a rate of 10 panels (80 holes) per 8-hour shift with a 20% time margin. Around-the-clock operations with 20 effective hours per day over a 9-month time period would require crews totaling 260 men for this function. The process of pouring the footings and emplacement of plates to which the rectenna panel structures are attached should take less time than excavation, but to maintain a continuous flow operation, it has been assumed that the times are identical.

The newer concrete trucks can deliver 10 cubic yards of mix per load. Given that an overall average requirement per footing is 6 cubic feet, then the trucks can supply enough mix to provide for 45 footings. In order not to detain the truck or a driver while footings are being poured, the concrete will be delivered from the mix plant to a mobile hopper at the work site. Hoses emanating from the hopper are operated by the two-man crews who are setting the footings. A turnaround cycle for the truck is estimated at 2-hours, thus in an 8-hour shift, a single truck can supply enough concrete to form the piers for 22.5 rectenna panels. If a truck is down for scheduled and unscheduled maintenance for 3-shifts out of the 21 during a week, then 30 10-cubic yard concrete trucks will be the complement required. If a mobile hopper supports the operations of two crews, then only 22 of these machines will be needed.

Rectenna Panel Assembly Concept

On-site assembly of the 436,818 rectenna panels presents the major construction time challenge. The large number dictates the need for multiple, semi-automated assembly equipments. The concept shown in Figure 9.9-5 fulfills that requirement. It is essentially a mobile construction jig which is assembled on-site and can be disassembled for transport to another rectenna site. The concept shown is initially loaded with materials for assembly of 10 rectenna panels. Since each rectenna panel weighs 2080 kilograms (4576 lbs), then a single flat-bed truck trip can deliver the 10 sets of rectenna panel materials. As each rectenna panel is completed, it is lifted from the construction jig by a truck crane (see insert on Figure) and set on concrete piers.

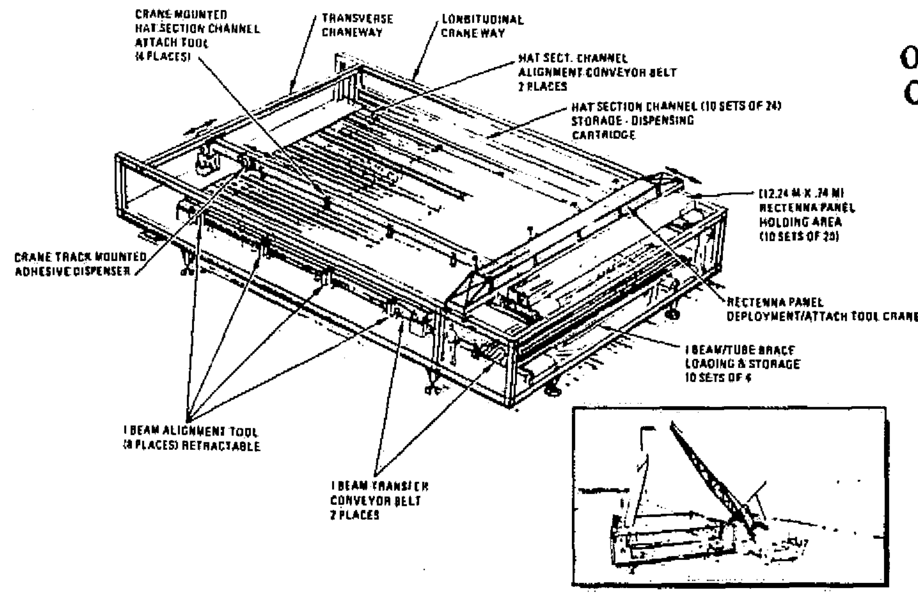


Figure 9.9-5. Rectenna Panel Assembly Concept

The panel assembly sequence is as follows: From the end and side of the jig where the sets of material have been loaded, the I-beams (with attached tube-braces) and hat sections are moved into place by conveyors with jig-stops to properly position each piece. After alignment is checked, a transverse craneway travels over the structural elements riveting the hat sections to the I-beam while applying an adhesive to the flanges. Next the longitudinal craneway lays down the rectenna panel modules onto the completed structural frame. Wiring harness hookups arethen made, a hoist sling from the truck crane is attached and the rectenna panel is removed from the construction jig. As the next set of hat-sections and I-beams are being conveyed into position, both craneways are returned to their initial positions.

The time sequence of these operations is shown in Figure 9.9-6. For equipment and crew size estimates, a time of 1-hour is assumed for the assembly of a rectenna panel. Under these assumptions, approximately 80 construction jigs would be required over a 9-month, 20-hour-effective day time

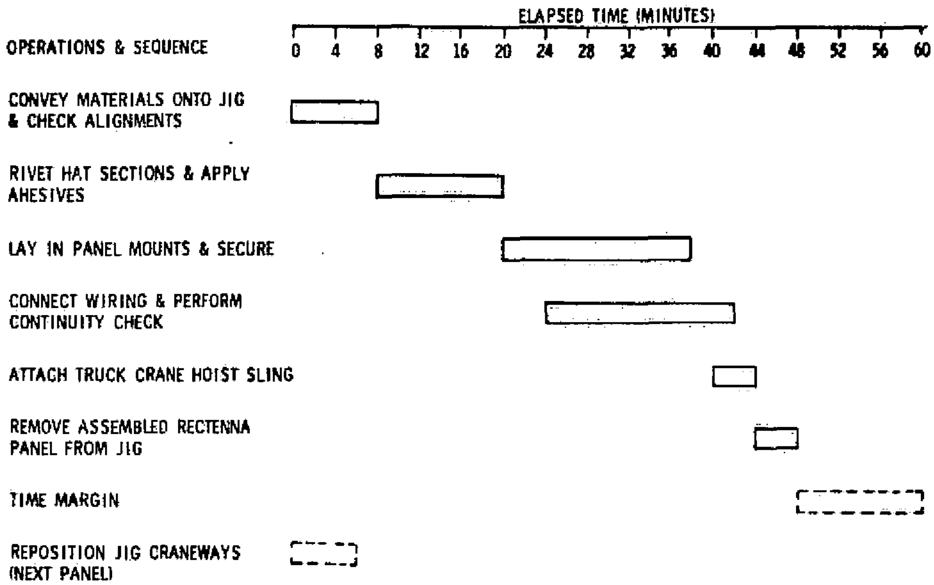


Figure 9.9-6. Rectenna Panel Assembly Timeline

period. With a crew size of three men per jig, approximately 2,430 man-months of labor are required. Installation of the completed rectenna panel on concrete piers is estimated to take about 20 minutes, therefore, one truck crane and a 3-man crew can support the operation of two construction jigs.

The primary purpose of postulating equipment and manpower needs is to ensure functional scheduling compatibilities and to develop realistic construction time estimates since supporting labor and hardware costs may ultimately prove to be less than 10% of materials costs. Estimates of time, major operations and labor for both site preparation and construction are shown in Figure 9.9-7.

9.9.2 RECTENNA SITE LOGISTICS

In order to meet the rectenna site construction schedule (Figure 9.9-8), construction masses must be supplied to the assembly and support equipments at rates which meet or exceed their demands. These mass flow demands - millions of kilograms per day - are depicted in Figure 9.9-8 by material type and as two types of demand: delivery to the site demands and intra-site demands. Delivery to site requirements are lower since pre-construction build up will allow, overall, approximately 12 months for satisfying these logistics demands. Intra-site requirements (for the same total masses) must be effected over a nine-month period. As noted, approximately 420 truck trips/day must be handled at the site. In terms of vehicle flow on a good highway, this is a relatively modest demand, but at the site, approximately 20 unloading docks will be required to handle the freight traffic. Unit trains of 100 cars may be cost-effective in a supplementary role, but the dominant masses must be handled by trucks if this schedule is to be maintained. Although the daily intra-site mass flow demands are higher, they are more easily handled inasmuch as a truck at the site can make a number of short trips per shift. Estimates have been made for the number of trucks and construction equipments required at the site as listed on the figure.

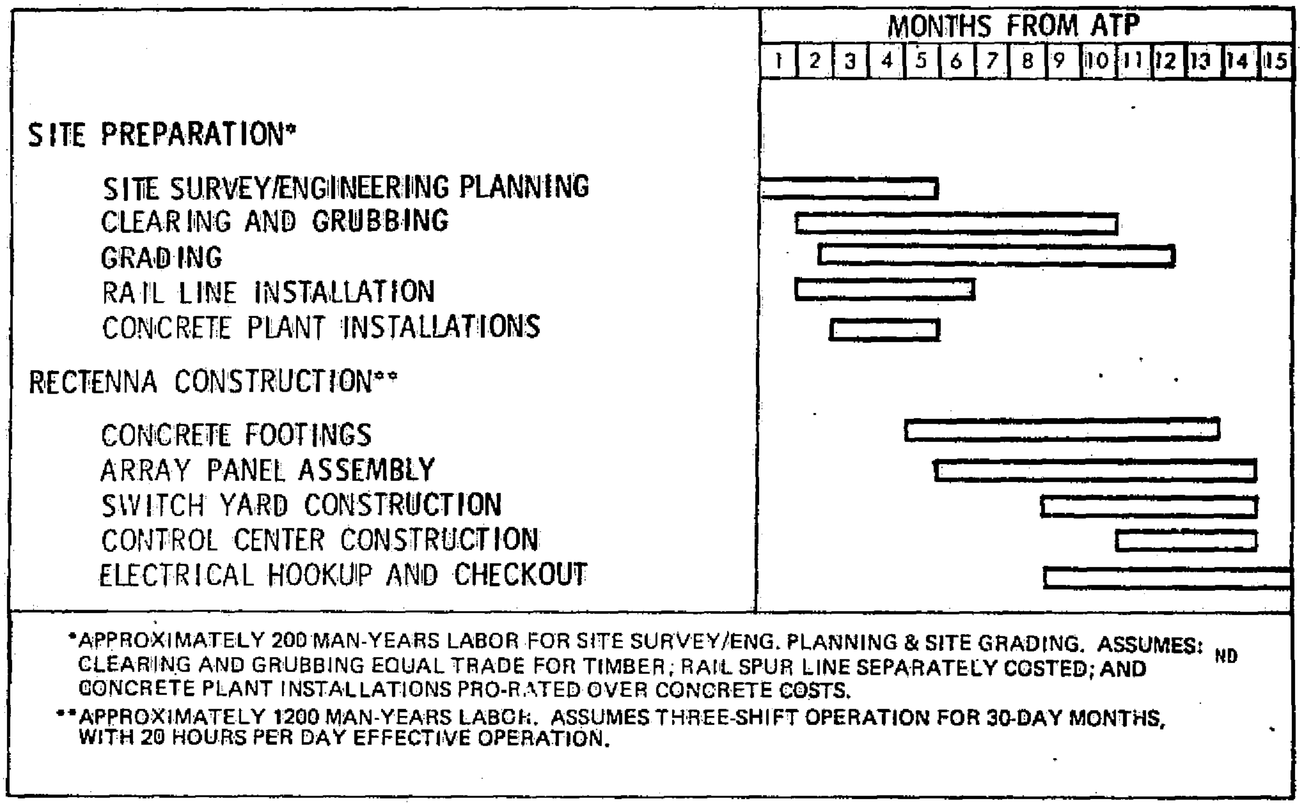
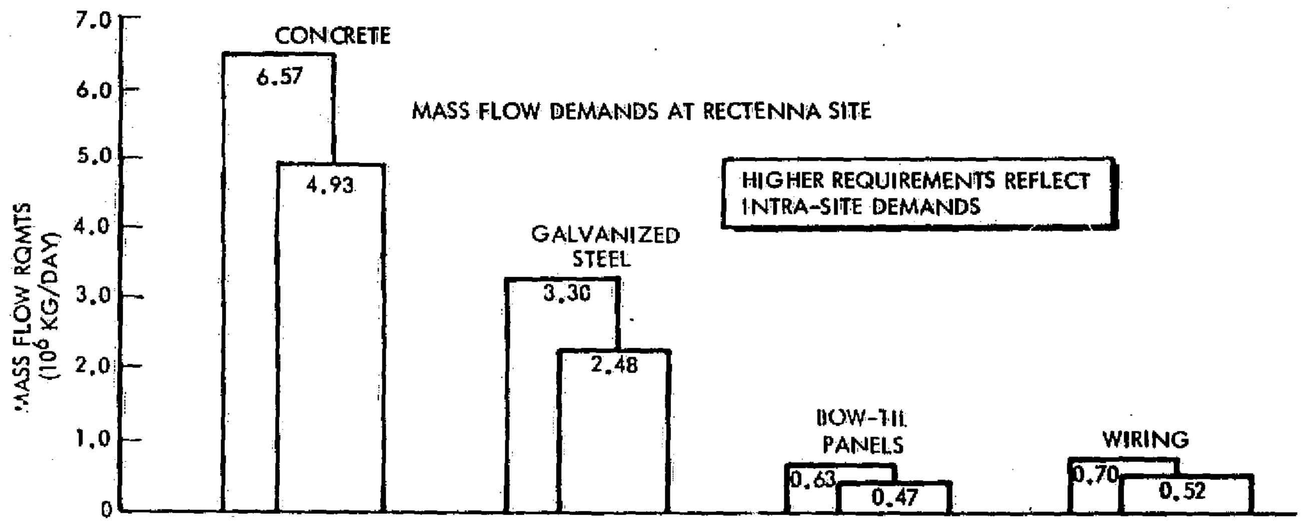


Figure 9.9-7. Rectenna Site Preparation & Construction Schedule



DELIVERY TO SITE REQUIREMENTS

- ~174 20-MTON CAPACITY FREIGHT TRUCK TRIPS PER DAY
- ~ 246 20-MTON CAPACITY SAND/GRAVEL TRUCK TRIPS PER DAY
- ~ 20 FREIGHT UNLOADING DOCKS AT SITE

INTRA-SITE REQUIREMENTS

- ~ 40 10-CUBIC YARD CONCRETE TRUCKS
- ~ 28 20-MTON CAPACITY FLAT BEDS

CONSTRUCTION EQUIPMENTS

- ~80 PANEL ASSEMBLY UNITS
- ~40 TRUCK CRANES

Figure 9.9-8. Rectenna Site Logistics

10.0 LEO-GEO VS. GEO SATELLITE CONSTRUCTION TRADES

10.0 LEO-GEO VS. GEO SATELLITE CONSTRUCTION TRADES

10.1 CONSTRUCTION ANALYSIS

Design integration of a 5-GW solar photovoltaic satellite concept which can be partially constructed in LEO has been completed and compared against an all-GEO constructed configuration. The two concepts are shown in Figure 10.1-1.

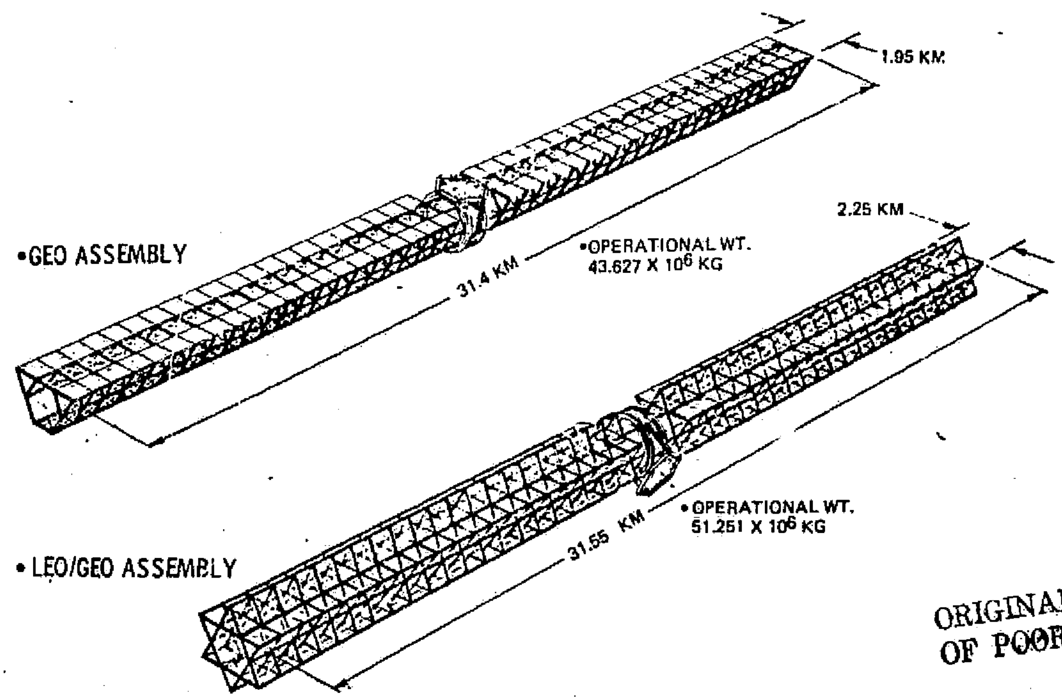


Figure 10.1-1. CR-1 Satellite Configurations
(LEO/GEO Assembly Vs. GEO Assembly)

As would be expected, a LEO-GEO constructed satellite must be oversized to accommodate the increased solar blanket areas required to offset the Van Allen Belt radiation damage incurred during transfer. Additionally, the two-order-of-magnitude increase in gravity gradient torques and resultant decrease in first-mode bending frequency required modifications to the structural configuration as shown. The operational weight increases, however, are predominantly due to the solar blanket mass oversize requirements and associated power distribution wiring mass. From the standpoint of construction operations, the LEO-GEO configuration presents a more complex assembly problem, and current structural design studies are expected to yield a more construction-preferred concept.

To achieve realistic weights LEO-GEO concept, for the electric OTV, and to acquire data needed for estimating ACS requirements (e.g., for lighted and eclipsed periods of the transfer), it was found necessary to simulate the LEO-to-GEO transfer on a computer program. Rockwell's GEOTOP II computer program (Figure 10.1-2) was used for electric OTV trajectory analysis. It is intended for low-thrust geocentric orbits, and can either average over each group of orbits or update the state equations several times per orbit. Solar array power degradation is computed at each trajectory step from Van Allen proton flux maps and energy-dependent GaAs damage factors for 50-μm (2-mil) cover thickness (2-MeV proton cutoff). Features used in this work include loss of power in the earth's shadow.

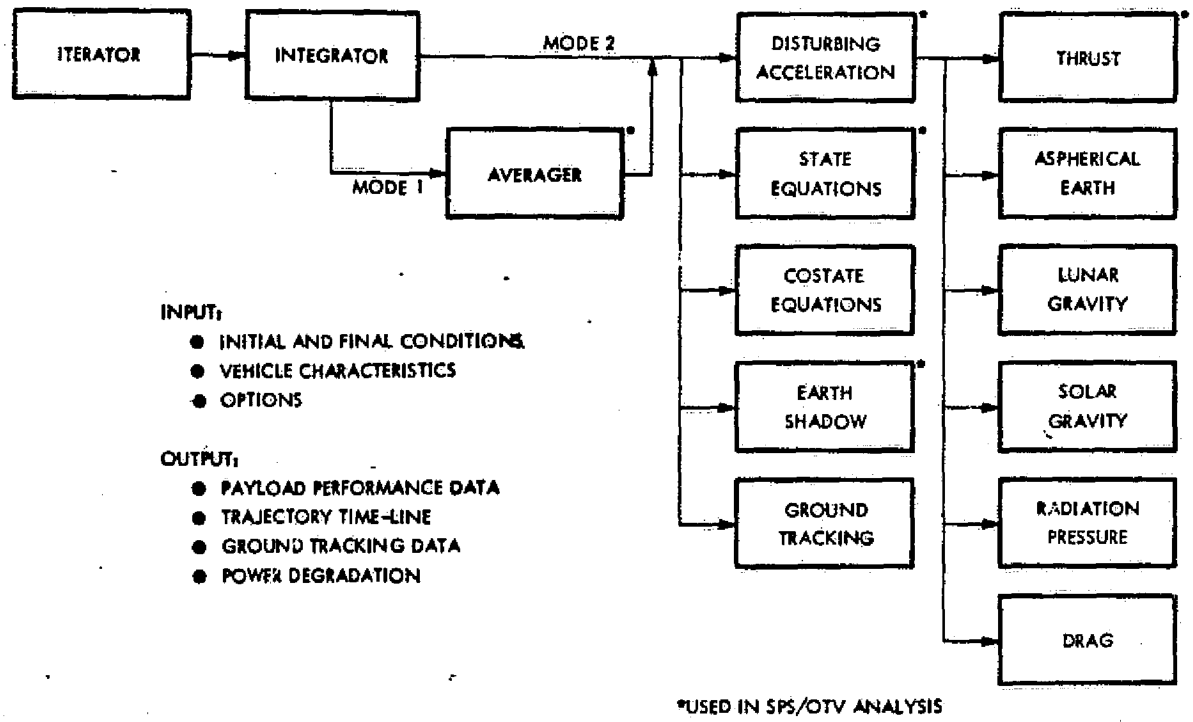


Figure 10.1-2. GEOTOP II Computer Program

Aerodynamic drag and solar radiation pressure can be included in GEOTOP II. These effects are small, however, for an SPS whose orbital altitude is 300 nmi and exposed area is primarily due to the solar blanket needed for propulsive power. In the case of the CR-1 photovoltaic SPS with a 176-day ascent to GEO, the drag force at 556 km (300 nmi) is 1250 N when the velocity is normal to the blanket surface. This is only 4 percent of the nominal electric thrust, and decreases to 1 percent at 664 km (360 nmi). The solar radiation pressure is 105 N (constant), or about 0.3 percent of the electric thrust.

The mass penalty for ACS is significant, however, even while holding the satellite attitude control to 0.1°. All-electric thruster ACS was deemed undesirable due to the weights incurred for employing battery generator power during the eclipse periods, therefore a storable chemical system (I_{sp} = 338) was selected for use during darkness. The resultant chemical ACS system was calculated to weigh 1.4×10^6 kg.

Another problem encountered was in uprighting or orienting the spacecraft after LEO construction has been completed. The high thrust levels required could not be achieved by reasonable numbers of electric thrusters, therefore a chemically fueled IOTV tug would be required with approximately 250×10^3 kg of propellant needed per satellite to perform this maneuver. When these mass requirements are totaled, approximately 60.445×10^6 kg will have to be launched from earth for the LEO-GEO-constructed satellite.

Analyses of these data indicated a heavy penalty for solar cell degradation and mass of a satellite with CR-1; therefore, a solar photovoltaic satellite with CR-2 (current point design) was investigated. The results of these trades in terms of cost differences are discussed below.

10.2 COST DIFFERENCES FOR GEO AND LEO/GEO CONSTRUCTION

Table 10.2-1 compares the "delta" costs for all construction in geosynchronous orbit (GEO) and for partial construction in low earth orbit (LEO) with completion of construction in GEO. Two OTV approaches are shown for all GEO construction: A LO_2/LH_2 chemical OTV and an electric OTV. For partial LEO and partial GEO construction, electric propulsion is also used, but the partially-constructed SPS provides the power to the electric thrusters and

Table 10.2-1. Cost Differences for GEO and LEO/GEO Construction

GaAlAs PHOTOVOLTAIC SATELLITE POINT DESIGN

	CHEMICAL OTV	ELECTRIC OTV	
	GEO CONSTRUCTION	LEO/GEO CONSTRUCT.	GEO CONSTRUCTION
NO. OF HLLV LAUNCHES	1092	451* (434)**	480 (453)
EARTH LAUNCH COSTS	$2597 x 10^6	$1026 x 10^6 ($988 x 10^6)	$1092 x 10^6 ($2029 x 10^6)
ELEC. PROP. MODULE REPLACEMENT COSTS	-	128 x 10^6 (128 x 10^6)	156 x 10^6 (156 x 10^6)
SOLAR BLANKET REPLACEMENT COSTS	-	46 x 10^6 (0)	522 x 10^6 (0)
SATELLITE OVERSIZING COSTS	-	202 x 10^6 (0)	- (-)
INTEREST COSTS (7.5%)	-	272 x 10^6 (272 x 10^6)	225 x 10^6 (225 x 10^6
TOTALS	$2597 x 10^6	$1674 x 10^6 ($1388 x 10^6)	$1995 x 10^6 ($1410 x 10^6)

*NOT SELF-ANNEALING
**SELF-ANNEALING

transports the mass to be used for partial GEO construction to GEO (self-propulsion). Since long-duration travel is involved through the Van Allen belt for both electric propulsion concepts (the electric OTV and self-propelled SPS), radiation damage to the solar cells was considered. The costs shown in parentheses assume that the GaAlAs solar cells are self-annealed to their original condition after passing through the radiation belt. The costs not in parentheses assume that the radiation damage is not annealed. Interests costs are included because of the long trip-time from LEO to GEO for electric propulsion transfer.

A comparison of the totals shows that GEO construction using a chemical OTV for cargo transfer from LEO to GEO costs about \$600 $\bar{M}$ more than GEO construction using an electric OTV (no cell annealing). When electric propulsion is used, GEO construction is about \$320 $\bar{M}$ greater than combined LEO/GEO construction without solar cell annealing. As indicated in this table, solar cell self-annealing results in virtually the same cost for combined LEO/GEO construction and all GEO construction.